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NASA Contractor Report 3394

Satellite Power Systems (SPS) Concept Definition Study (Exhibit D)

Volume III - Transportation Analysis

G. M. Hanley

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Volume III - Transportation Analysis

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FOREWORD

Volume III, *Transportation Analyses*, of the SPS Concept Definition Study final report is submitted by Rockwell International through the Space Operations and Satellite Systems Division. All work was completed in response to NASA/MSFC Contract NAS8-32475, Exhibit D.

The SPS final report provides the NASA with additional information on the selection of a viable SPS concept, and furnishes a basis for subsequent technology advancement and verification activities. Other volumes of the final report are listed below.

Volume	Title
I	Executive Summary
II	Systems/Subsystems Analyses
IV	Operations Analyses
V	Systems Engineering/Integration Research and Technology
VI	Cost and Programmatics

The SPS Program Manager, G. M. Hanley, may be contacted on any technical or management aspects of this report. He can be reached at (213) 594-3911, Seal Beach, California.

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GLOSSARY

A	Ampere
Å	Angstrom
ac	Alternating current
ACSS	Attitude control and stationkeeping system
AMO	Air mass zero
ARDS	Attitude reference determination system
$\overline{B}$	Billions of dollars
B_eO	Beryllium oxide (Berlox)
BCD	Binary coded decimal
BCU	Bus control units
BOL	Beginning of life
BT	Battery tie contactor
°C	Degree centigrade
C_e	Cesium
cm	Centimeter
CMD	Command
COTV	Cargo orbital transfer vehicle
CPU	Central processing unit
CR	Concentration ratio
CR_E	Effective concentration ratio
CVD	Controlled vapor deposit
D/A	Digital to analog
dB	Decibel
dc	Direct current
DOE	Department of Energy
DVM	Digital voltmeter

EBS	Electron beam semiconductor
E_g	Bandgap energy
EMI	Electromagnetic interference
EOL	End of life
EOTV	Electric orbital transfer vehicle
EVA	Extra-vehicular activity
f	Frequency
°F	Degree Fahrenheit
FEP	Adhesive material
FET	Field-effect transistor
FOC	Final operational capability
f_p	Pilot frequency
f_r	Reference signal frequency
f_T	Transmitted frequency
G	Giga- (10^9)
G	Gear, switch
GaAlAs	Gallium aluminum arsenide
GaAs	Gallium arsenide
GEO	Geosynchronous, equatorial orbit
GHz	Gigahertz
GPS	Global Positioning System
GRS	Ground receiving station
GW	Gigawatt
HLLV	Heavy-lift launch vehicle
HPWB	Half-power-point beamwidth
HV	High voltage
Hz	Hertz
IB	Interface bus
IBM	International Business Machines Corp.
IMCS	Information management and control system
IMS	Information management system (see IMCS)

IOC	Initial operations capability
IOP	In-orbit plane
IOTV	Inter-orbit transfer vehicle
IUS	Inter-orbit utility stage
k	Kilo (10^3)
K	Potassium
°K	Degree Kelvin
km	Kilometer (1000 meters)
kN	Kilonewton
KSC	Kennedy Space Flight Center
kV	Kilovolts
LED	Light-emitting diode
LEO	Low earth orbit
LH_2	Liquid hydrogen
LOX	Liquid oxygen
LPE	Liquid phase epitaxal
LRB	Liquid rocket booster
LRU	Line replaceable unit
LSST	Large space structures technology
m	Meter
M	Mega- (10^6)
MBG	Multi-bandgap
MC-ABES	Multi-cycle airbreathing engine system
MeV	Millions of electron volts
μp	Microprocessor
MPCA	Master phase reference control amplifier
MPTS	Microwave power transmission system
MSFC	Marshall Space Flight Center
MTBF	Mean time between failure
MTTF	Mean time to failure
MW	Megawatt
MW	Microwave

MW_e	Megawatt—electrical
MWM	Manned work modules
MW_T	Megawatt—thermal
M_x	Disturbance torque along X-axis
N	Newton
NaK	Sodium-potassium
NASA	National Aeronautics and Space Administration
N-S	North-South
O&M	Operations and maintenance
OTV	Orbit transfer vehicle
PDS	Power distribution system
PLV	Personnel launch vehicle
PM	Personnel module
POP	Perpendicular to orbit plane
POTV	Personnel orbital transfer vehicle
psi	Pounds per square inch
RAC	Remote acquisition and control
R&D	Research and development
R&T	Research and technology
RCA	Radio Corporation of America
RCI	Replacement cost investment
RCR	Resonant cavity radiator
RCS	Reaction control system
RF	Radio frequency
RFI	Radio frequency interference
RTE	Real-time evaluation
S/A	Solar array
SCB	Space construction base
SG	Switch gear
Si	Silicon

SIT	Static induction transistor
SM	Sub-multiplexer
SOC	Space Operations Center
SPS	Satellite Power Systems
SRB	Solid rocket booster
STS	Space Transportation System
T	Temperature
TBD	To be determined
T&E	Test and evaluation
TFU	Theoretical first unit
TT&C	Telemetry, tracking, and communications
TWT	Traveling wave tubes
UI	Utility interface
V	Volt
VHF	Very high frequency
VSWR	Voltage standing wave ratio
VTO	Vertical take-off
W	Watt
Wh	Watt-hour
X,Y,Z	Coordinate axes of satellite

Symbols

ε	Error signals
λ	Wavelength of frequency f (Hertz)
μ	Micro-
η	Efficiency
ϕ	Phase
ϕ	Coordinate axis angle—Phi
θ	Coordinate axis (angle)—Theta

1.0 INTRODUCTION

During the several phases of the SPS Concept Definition Study, various transportation system elements were synthesized and evaluated on the basis of their potential to satisfy overall SPS transportation requirements and of their sensitivities, interfaces, and impact on the SPS.

Additional analyses and investigations were conducted to further define transportation system concepts that will be needed for the developmental and operational phases of an SPS program. To accomplish these objectives, transportation systems such as the Shuttle and its derivatives have been identified; new heavy-lift launch vehicle (HLLV) concepts, cargo and personnel orbital transfer vehicles (EOTV and POTV), and intra-orbit transfer vehicle (IOTV) concepts have been evaluated; and, to a limited degree, the program implications of their operations and costs were assessed. The results of these analyses have been integrated into other elements of the overall SPS concept definition studies.

The primary areas of study during this phase of the contract were directed toward the following:

- The synthesis and evaluation of a smaller payload version of the HLLV
- The assessment of specific technical issues relating to HLLV feasibility
- A reassessment of the EOTV concept and configuration update
- The identification of technology advancment requirements to enhance/satisfy operations requirements
- The generation of cost and programmatic data to support SPS concept trade studies

SPS program and transportation system analyses continue to show that a prime element of transportation systems cost, and SPS program cost, is that of payload delivery to LEO or HLLV feasibility/cost.

2.0 TRANSPORTATION SYSTEM ELEMENTS

Studies conducted to date definitely show that the SPS program will require a dedicated transportation system. In addition, because of the high launch rate requirements and environmental considerations, a dedicated launch facility may also be required during the SPS construction phase. The major elements of the SPS transportation system consist of:

- Heavy-Lift Launch Vehicle (HLLV)—SPS cargo to LEO
- **Personnel Launch Vehicle (PLV)—personnel to LEO (Growth STS)**
- Electric Orbit Transfer Vehicle (EOTV)—SPS cargo to GEO
- Personnel Orbit Transfer Vehicle (POTV)—personnel from LEO to GEO
- Personnel Module (PM)—personnel carrier from earth-LEO-GEO
- Intra-Orbit Transfer Vehicle (IOTV)—On-orbit transfer of cargo/personnel

Of the many HLLV options investigated (i.e., one- and two-stage ballistic or winged, parallel or series burn, etc.), a two-stage vertical takeoff horizontal landing (VTO/HL) HLLV (Figure 2.0-1) has been tentatively selected as the preferred or "baseline" concept. An interim HLLV will be required during

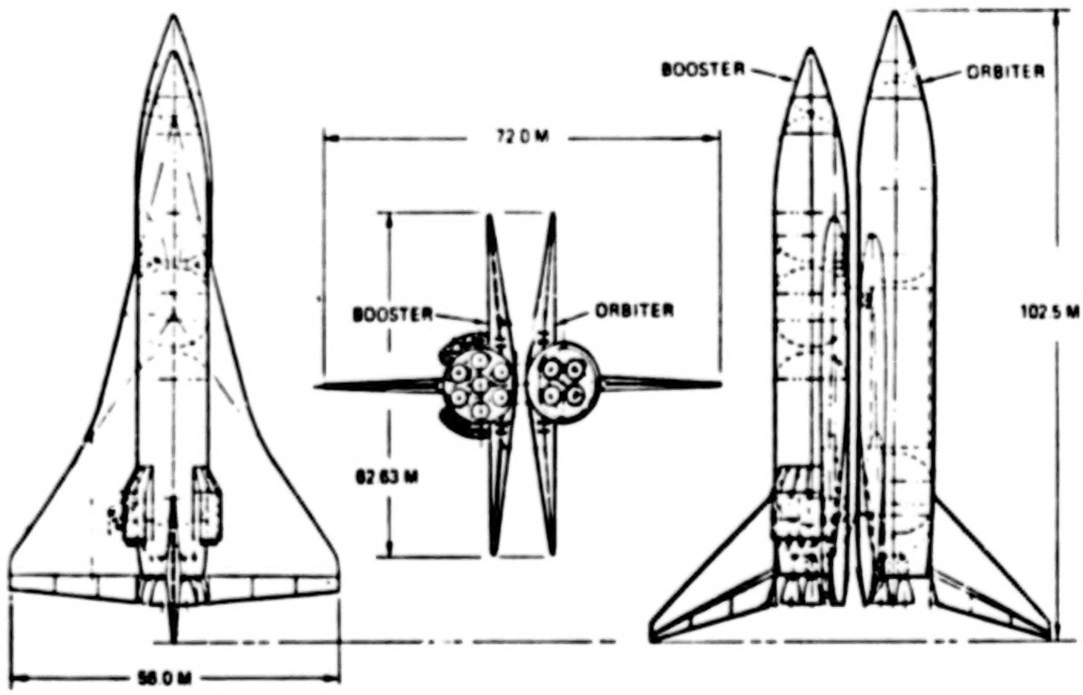


Figure 2.0-1. VTO/HL HLLV Concept

the initial SPS program development phase (Figure 2.0-2). This vehicle is designated as a Shuttle-derived or "Growth Shuttle" HLLV (STS-HLLV). This launch vehicle utilizes the same elements as the PLV (described below), except

the orbiter is replaced with a payload module and an auxiliary recoverable engine module to provide a greater cargo capability.

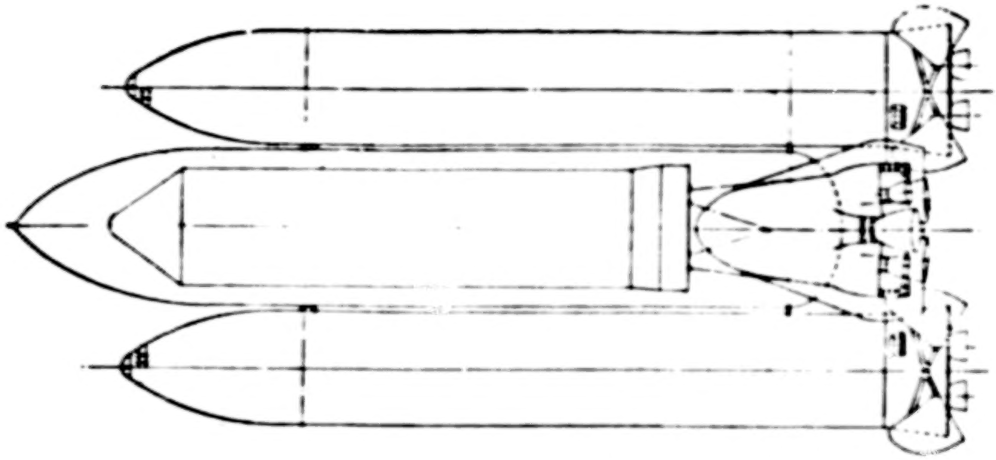


Figure 2.0-2. STS-HLLV Configuration

The personnel launch vehicle (PLV) is used to transfer the SPS construction crew from earth to LEO. This launch vehicle is a modified Shuttle Transportation System (STS) configuration. The existing STS solid rocket boosters (SRB) are replaced with reusable liquid rocket boosters (LRB), thus affording a greater payload capability and lower overall operating cost (Figure 2.0-3). The personnel module, described below, is designed to fit within the existing STS orbiter cargo bay.

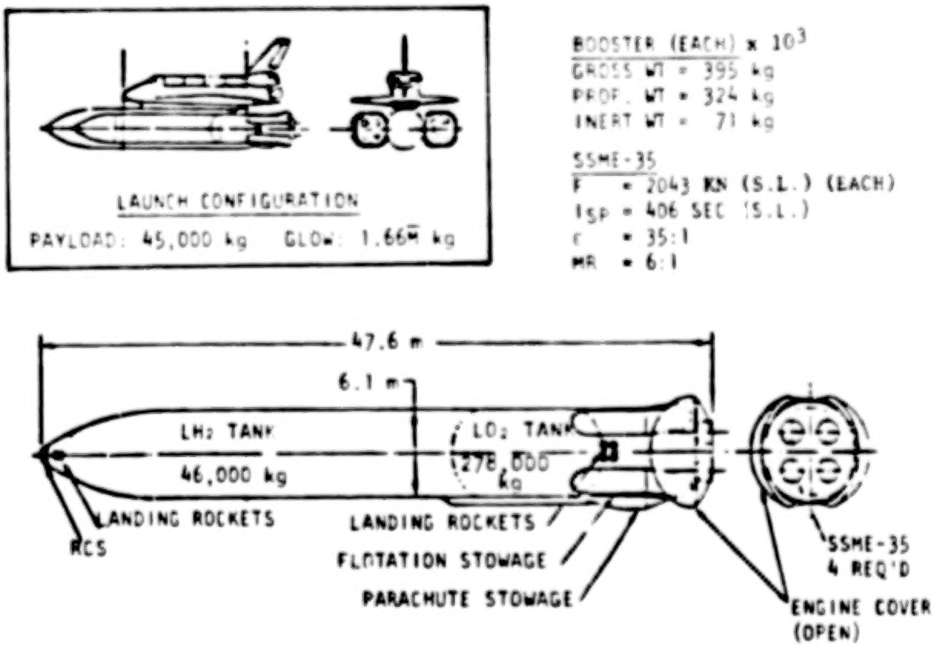


Figure 2.0-3. Growth Shuttle PLV

The interim HLLV and PLV (STS derivatives) will be phased out of the program when the SPS dedicated HLLV becomes operational.

The electric orbital transfer vehicle (EOTV) is employed as the primary transportation element for SPS cargo from LEO to GEO. The vehicle configuration (Figure 2.0-4), defined to accomplish this mission phase, utilizes the same power source and construction techniques as the SPS. The solar array consists of two "bays" of the SPS, electric argon ion engine arrays, and the requisite propellant storage and power conditioning equipment. The vehicle configuration, payload capability, and "trip time" have been established on the basis of overall SPS compatibility.

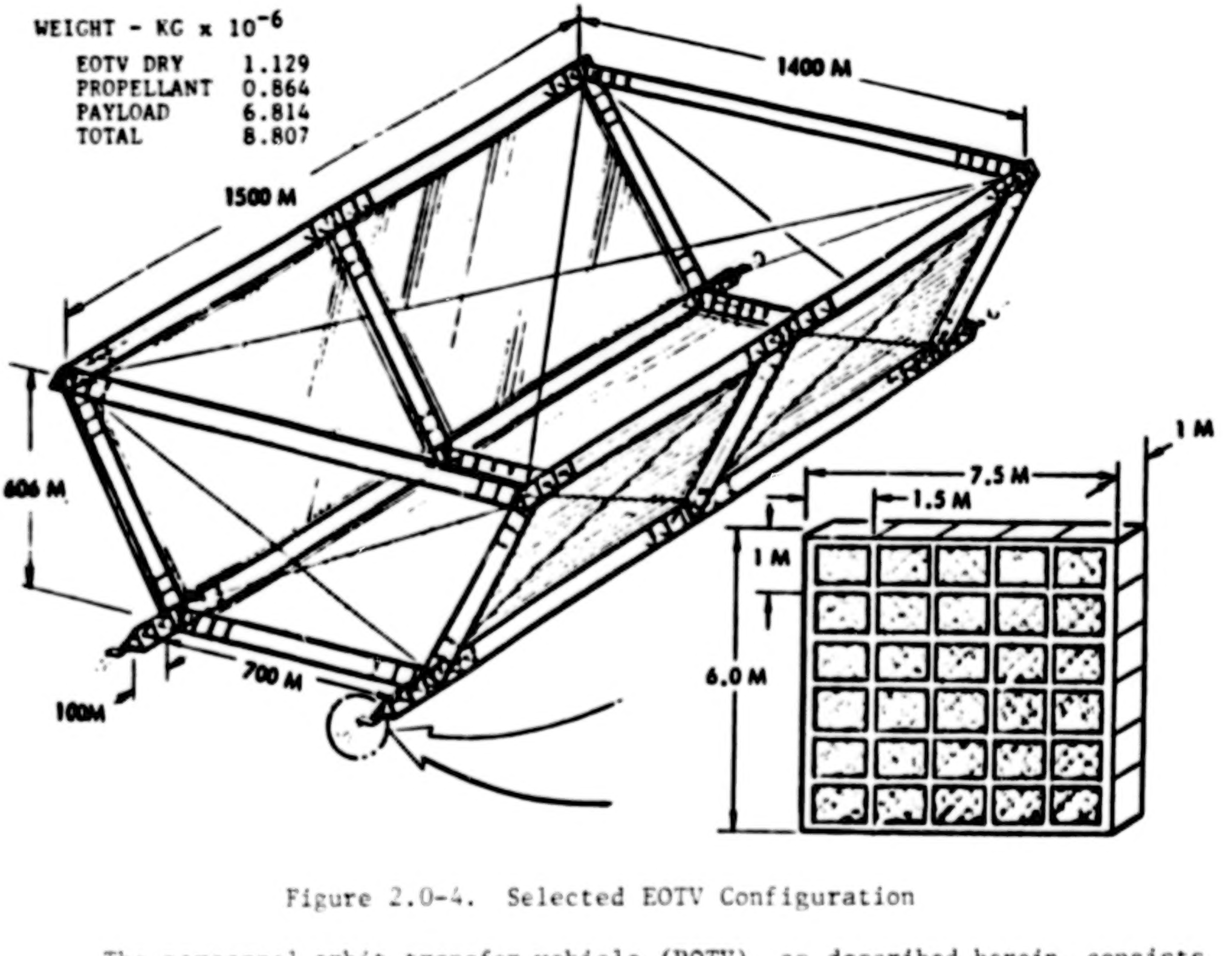
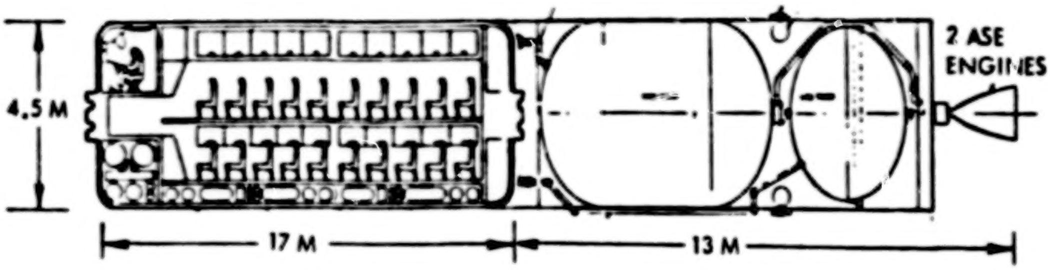


Figure 2.0-4. Selected EOTV Configuration

The personnel orbit transfer vehicle (POTV), as described herein, consists of that propulsive element required to transfer the personnel module (PM) and its crew/construction personnel from LEO to GEO. The mated configuration of POTV/PM is depicted in Figure 2.0-5. The POTV consists of a single, chemical (LOX/LH_2) rocket stage which is initially fueled in LEO and refueled in GEO for return to LEO. The POTV has been sized such that it is capable of fitting within the existing STS cargo bay and the growth STS payload delivery capability.

An intra-orbit transfer vehicle (IOTV) is defined in concept only. Because of the potential problems associated with docking and cargo transfer between the

HLLV and EOTV in LEO and the EOTV and GEO construction base, a transfer vehicle capable of accomplishing this function is postulated. From cost and programmatic aspects of the overall SPS program, this element is depicted as a chemical rocket stage, manned or remotely operated.



• 60 MAN CREW MODULE	18,000 KG
• SINGLE STAGE OTV (GEO REFUELING)	36,000 KG
• BOTH ELEMENTS CAPABLE OF GROWTH STS LAUNCH	

Figure 2.0-5. POTV Configuration

3.0 TRANSPORTATION SYSTEM SCENARIO

As previously stated, the SPS will require a dedicated transportation system. In addition, because of the high launch rates and certain environmental considerations, it appears that a dedicated launch facility may also be required for SPS HLLV launches. Transportation system LEO operations are depicted in Figure 3.0-1. The SPS HLLV delivers cargo and propellants to LEO, which are transferred to a dedicated electric OTV (EOTV) by means of an intra-orbit transfer vehicle (IOTV) for subsequent transfer to GEO.

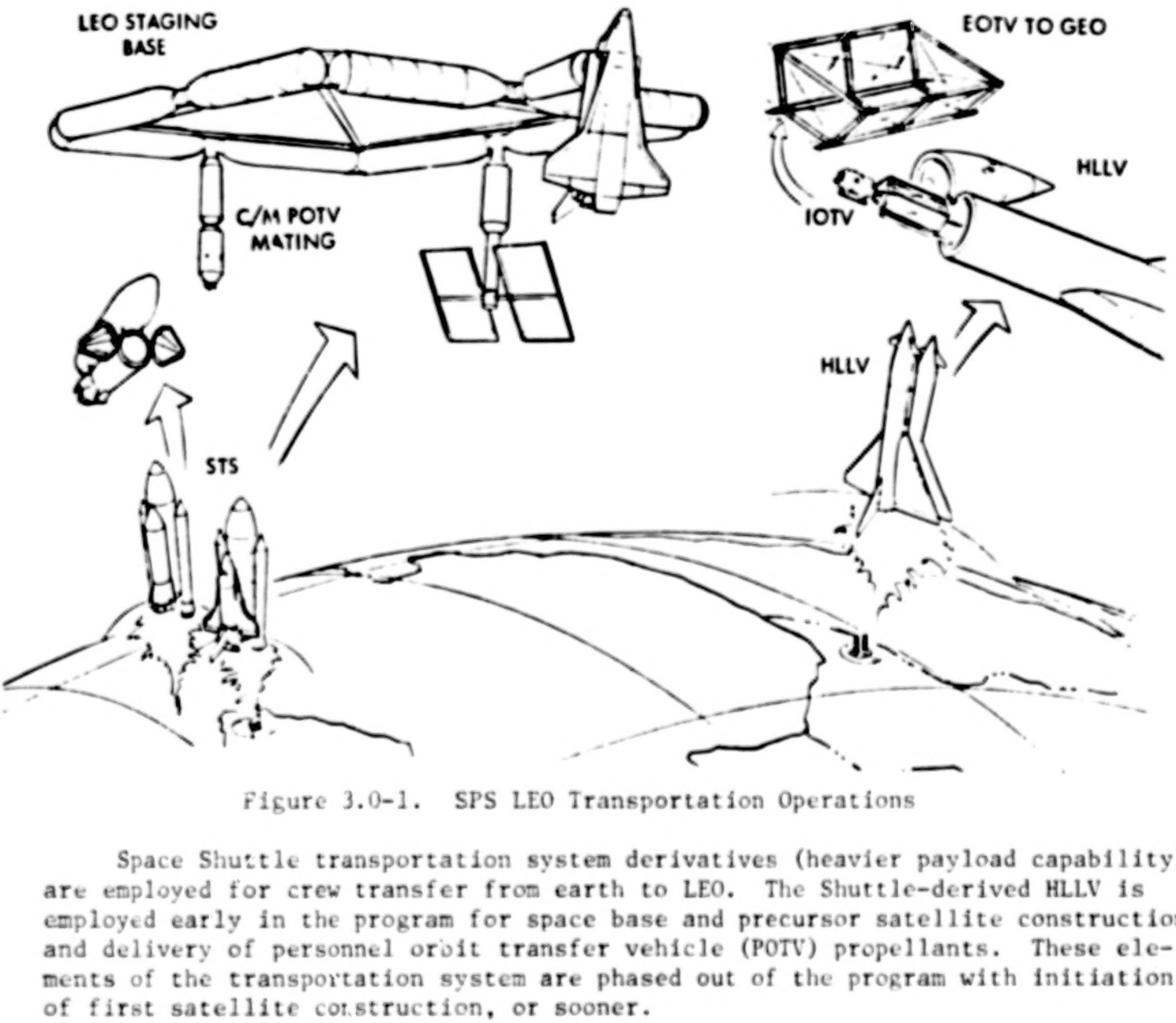


Figure 3.0-1. SPS LEO Transportation Operations

Space Shuttle transportation system derivatives (heavier payload capability) are employed for crew transfer from earth to LEO. The Shuttle-derived HLLV is employed early in the program for space base and precursor satellite construction and delivery of personnel orbit transfer vehicle (POTV) propellants. These elements of the transportation system are phased out of the program with initiation of first satellite construction, or sooner.

Transportation system GEO operations are depicted in Figure 3.0-2. Upon arrival at GEO, the SPS construction cargo is transferred from the EOTV to the SPS construction base by IOTV. The POTV with crew module docks to the construction base to effect crew transfer and POTV refueling for return flight to LEO. Crew consumables and resupply propellants are transported to GEO by the EOTV.

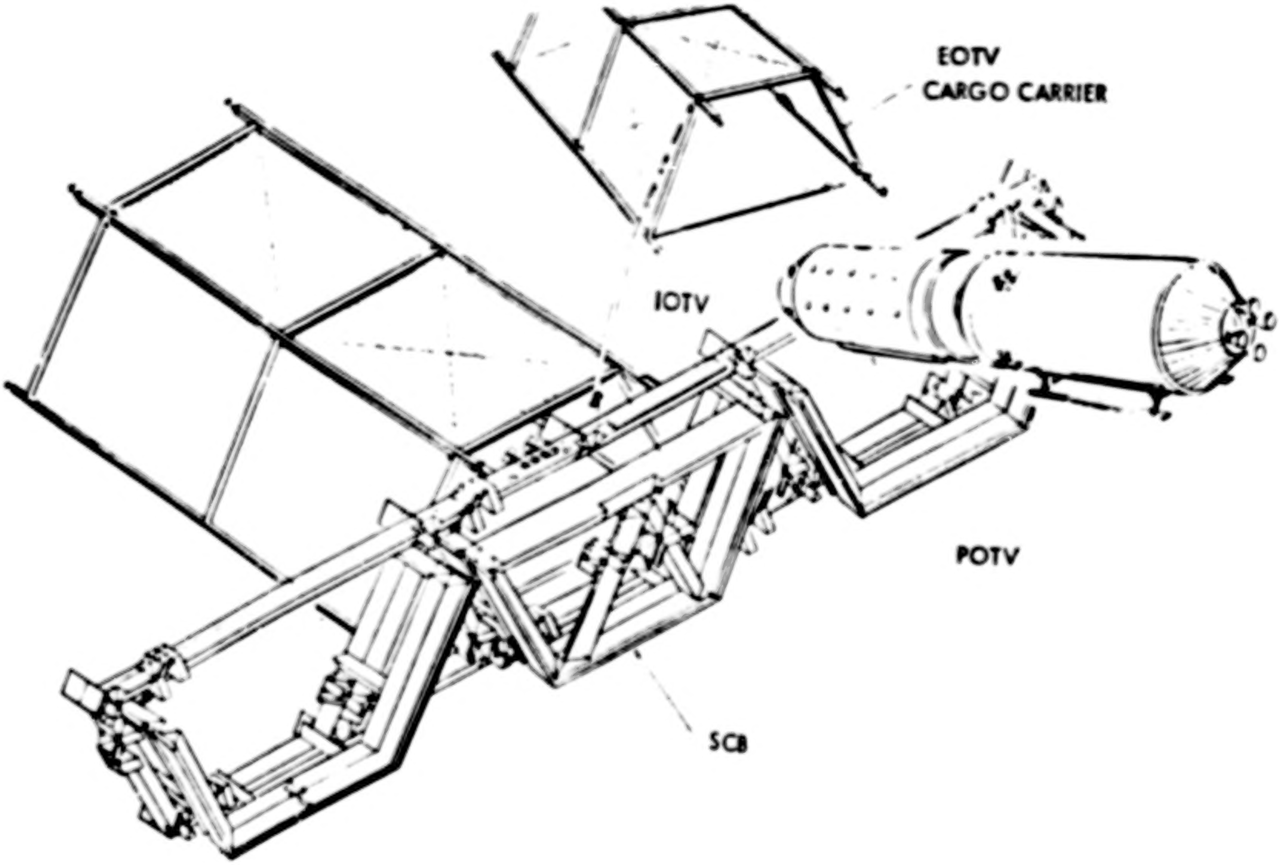


Figure 3.0-2. SPS GEO Transportation Operations

Transportation system requirements are dominated by the vast quantity of materials to be transported to LEO and GEO. Tables 3.0-1, -2, and -3 summarize the mass delivery requirements and numbers of vehicle flights for the reference satellite and transportation elements. All mass figures include a 10% packaging factor. Table 3.0-1 summarizes transportation requirements for construction

Table 3.0-1. GaAs Reference SPS Concept—Precursor Transportation Requirements

		VECHICLE FLIGHTS			
	MASS-10^6 kg	STS (PLV)	STS (CARGO)	STS-GROWTH (PLV)	STS-HLLV (CARGO)
PRECURSOR	2.019	6	79	-	-
LEO BASE	5 MODULES	-	-	-	5
SCB	5.300	-	-	72	58
PROPELLANT	0.864	-	34	-	-
TOTAL	-	6	113	72	63

of the precursor satellite. Table 3.0-2 is a summary of requirements for first satellite construction. Table 3.0-3 defines the transportation requirements during the total 60 year program. The average annual mass to LEO during the construction phase is in excess of 100 million kilograms with more than 400 HLLV launches per year.

Table 3.0-2. GaAs Reference SPS Concept—
TFU Transportation Requirements

	MASS - 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE CONSTR. & MAINT.	34.8	34.8	5.4	153.3	40	5.1	215	153
CREW CONSUMABLES	1.5	0.1		6.6		-	7	-
POTV PROPELLANTS	2.9	1.4		12.7		0.2	13	6
EOTV CONSTRUCTION & MAINT.	7.5	-		32.8		-	33	-
EOTV PROPELLANTS	7.6	-		33.5		-	34	-
IOTV PROPELLANTS	0.2	0.1		0.6		-	1	1
SCB TO GEO	-	-		-		2	-	-
TOTAL	54.5	36.4	5	240	40	8	303	160
FLEET	-	-	-	5	4	6	2	2

Table 3.0-3. GaAs Reference SPS Concept—
Total Transportation Requirements, 60-Year Program
(60 Satellites)

	MASS - 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE								
CONSTRUCTION	2087.7	2087.7	111	9,197	1220	306.4	10,741	9,197
OPS & MAINT.	492.2	492.2	34	2,168	324	72.7	2,560	2,168
CREW CONSUMABLES								
CONSTRUCTION	29.9	28.7		132		4.2	132	126
OPS & MAINT.	9.2	7.6		41		1.1	41	34
POTV PROPELLANTS								
CONSTRUCTION	87.9	44.0		387		6.5	[illegible]	194
OPS & MAINT.	23.3	11.7		103		1.7	103	52
EOTV CONSTRUCTION								
CONSTRUCTION	19.9	12.4		88		1.8	88	55
OPS & MAINT.	5.0	5.0		22		0.7	22	22
EOTV PROPELLANTS								
CONSTRUCTION	306.0	1.9		1,348		0.3	1,348	8
OPS & MAINT.	73.0	0.8		322		0.1	322	4
IOTV PROPELLANTS								
CONSTRUCTION	7.4	3.2		33		0.5	33	14
OPS & MAINT.	1.8	0.8		8		0.1	8	3
SUMMARY								
CONSTRUCTION	2538.8	2177.9	111	11,185	1220	320	12,729	9,594
OPS & MAINT.	604.5	518.1	34	2,664	324	76	3,056	2,283
TOTAL	3143.3	2696.0	145	13,849	1544	396	15,785	11,877
VEHICLE FLEET								
CONSTRUCTION	-	-	-	38	12	16	112	
OPS & MAINT.	-	-	-	9	3	4	27	
TOTAL	-	-	-	47	15	20	139	

4.0 HEAVY-LIFT LAUNCH VEHICLE

Evolving Satellite Power System (SPS) program concepts envision the assembly and operation of 60 solar-powered satellites in synchronous equatorial orbit over a period of 30 years. With each satellite weighing approximately 35 million kilograms, economic feasibility of the SPS is strongly dependent upon low-cost transportation of SPS elements to LEO. The rate of delivery of SPS elements alone to LEO for this projected program is 70 million kilograms per year. This translates into as many as 350 flights per year, or one flight per day, using a fleet of vehicles, each delivering a cargo of 200,000 kg.

The magnitude and sustained nature of this advanced space transportation program concept require long-term routine operations somewhat analogous to commercial airline/airfreight operations. Ballistic vertical-takeoff, heavy-lift launch vehicles (e.g., 400,000-kg payload) can reduce the launch rate to 200 flights per year. However, requirements such as water recovery of stages with subsequent refurbishment, stacking, launch pad usage, and short turnaround schedules introduce severe problems for routine operations. The focus of attention has, therefore, been influenced in the direction of winged recoverable vehicle concepts.

A two-stage, vertical-takeoff/horizontal-landing, heavy-lift launch vehicle (VTO/HL HLLV) concept has been evaluated as a candidate for SPS cargo and personnel transport to low earth orbit (LEO). Two vehicle payload capability options were synthesized—one with a payload capability of approximately 227,000 kg (500,000 lb) during the Exhibit C contract effort, and the other 113,500 kg (250,000 lb) during the Exhibit D contract effort. Basic ground rules and assumptions employed in vehicle sizing are summarized in Table 4.0-1. Both stages have flyback capability to the launch site; the second **stage** is recovered in the same manner as the Shuttle Transportation System (STS) orbiter.

Table 4.0-1. HLLV Sizing—Ground Rules/Assumptions

- TWO-STAGE VERTICAL TAKEOFF/HORIZONTAL LANDING (VTO/HL)
- FLY BACK CAPABILITY BOTH STAGES - ABES FIRST STAGE ONLY
- PARALLEL BURN WITH PROPELLANT CROSSFEED
- LOX/RP FIRST STAGE - LOX/LH_2 SECOND STAGE
- HI P_c GAS GENERATOR CYCLE ENGINE - FIRST STAGE [I_s (VAC) = 352 SEC.]
- HI P_c STAGED COMBUSTION ENGINE - SECOND STAGE [I_s (VAC) = 466 SEC.]
- STAGING VELOCITY - HEAT SINK BOOSTER COMPATIBLE
- CIRCA 1990 TECHNOLOGY BASE - BAC/MMC WEIGHT REDUCTION DATA
- ORBITAL PARAMETERS - 487 KM @ 31.6°
- THRUST/WEIGHT - 1.30 LIFTOFF/3.0 MAX
- 15% WEIGHT GROWTH ALLOWANCE/0.75% ΔV MARGIN

The vehicle utilizes a parallel burn mode with propellant cross-feed from the first-stage tanks to the second-stage engines. The first stage employs high chamber pressure gas generator cycle LOX/RP fueled engines with LH_2 cooling, and the second stage employs a staged combustion engine similar to the Space Shuttle main engine (SSME) which is LOX/LH_2 fueled.

Although trade studies were conducted, a vehicle staging velocity compatible with a heat sink booster concept is considered desirable from an operations standpoint. Technology growth consistent with the 1990 time period was used to estimate weights and performance. The expected technology improvements are summarized in Table 4.0-2. Orbital parameters are consistent with SPS LEO base requirements, and the thrust-to-weight limitations are selected to minimize engine size and for crew/passenger comfort. Growth margins of 15% in inert weight and 0.75% in propellant reserves were established.

Table 4.0-2. Technology Advancement

• Body structure	17%
• Wing structure	15%
• Vertical tail	18%
• Canard	12%
• Thermal protection system	20%
• Avionics	15%
• Environmental control	15%
• Reaction control system	15%
• Rocket engines	
1st stage thrust/weight = 120	
2nd stage thrust/weight = 80	

HLLV performance was determined by the use of a modified STS scaling and trajectory program. The engine performance parameters used in the analysis are given in Table 4.0-3.

Table 4.0-3. Engine Performance Parameters

ENGINE	SPECIFIC IMPULSE (SEC)		MIXTURE RATIO	THRUST/WEIGHT
	SEA LEVEL	VACUUM		
LOX/RP GG CYCLE	329.7	352.3	2.8:1	120
LOX/CH_4 GG CYCLE	336.9	361.3	3.5:1	120
LOX/LH_2 STAGED COMB.	337.0	466.7	6.0:1	80

In addition to pertinent trade studies (i.e., propellant type and loading, engine throttling, staging velocity, etc.) several technical issues were addressed; these included vehicle flight characteristics, ascent control analyses, thrust load distribution and structural requirements, and a preliminary thermal/structural assessment. The latter studies were performed with the lighter payload HLLV option.

4.1 EXHIBIT C REFERENCE HLLV CONCEPT

The Exhibit C reference configuration is shown in Figure 4.1-1 in the launch configuration. As shown, both stages have common body diameter, wing, and vertical stabilizer; however, the overall length of the second stage (orbiter) is approximately 5 m greater than the first stage (booster). The vehicle gross liftoff weight (GLOW) is 7 million kg with a payload capability of 230 thousand kg to the reference earth orbit. A summary weight statement is given in Table 4.1-1. The propellant weights indicated are total loaded propellant (i.e., not usable). The second-stage weight (ULOW) includes the payload weight. During the booster ascent phase, the second-stage LOX/LH_2 propellants are cross-fed from the booster to achieve the parallel burn mode. Approximately 730 thousand kg of propellant are cross-fed from the booster to the orbiter during ascent.

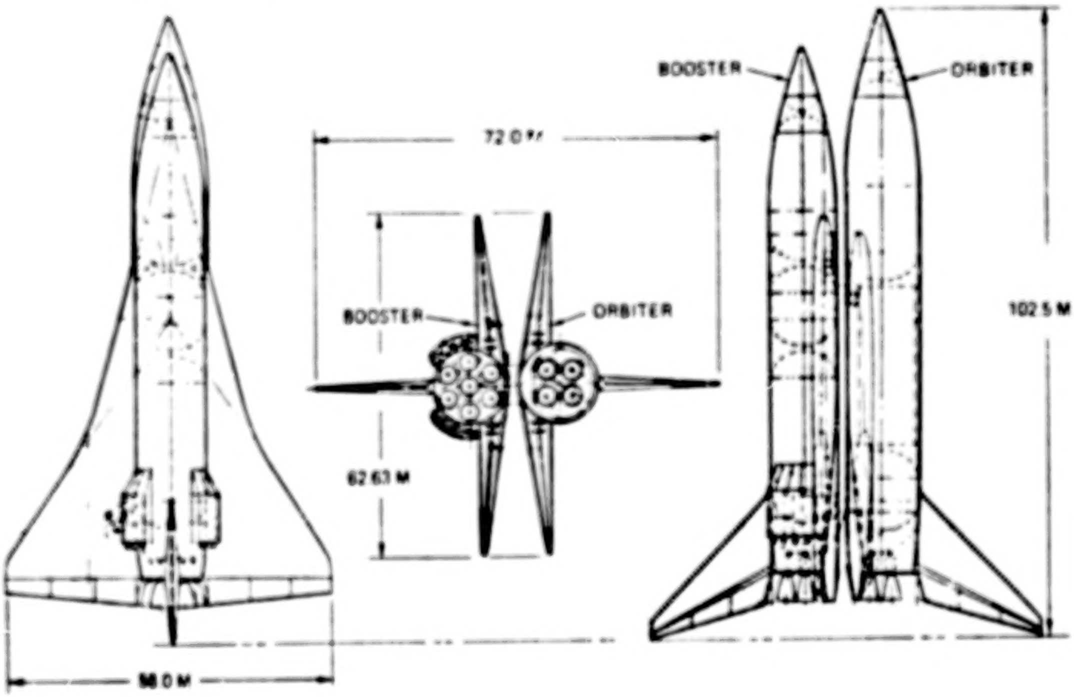


Figure 4.1-1. Reference HLLV Launch Configuration

Table 4.1-1. HLLV Mass Properties ($\times10^{-6}$)

	kg	lb
GLOW	7.14	15.73
BLOW	4.92	10.84
W_{P1}	4.49	9.89
ULOW	2.22	4.89
W_{P2}	1.66	3.65
PAYLOAD	0.23	0.51

The HLLV booster, shown in the landing configuration in Figure 4.1-2, is approximately 90 m in length with a wing span of 56 m and a maximum clearance height of 35 m; the nominal body diameter is 18 m. The vehicle has a

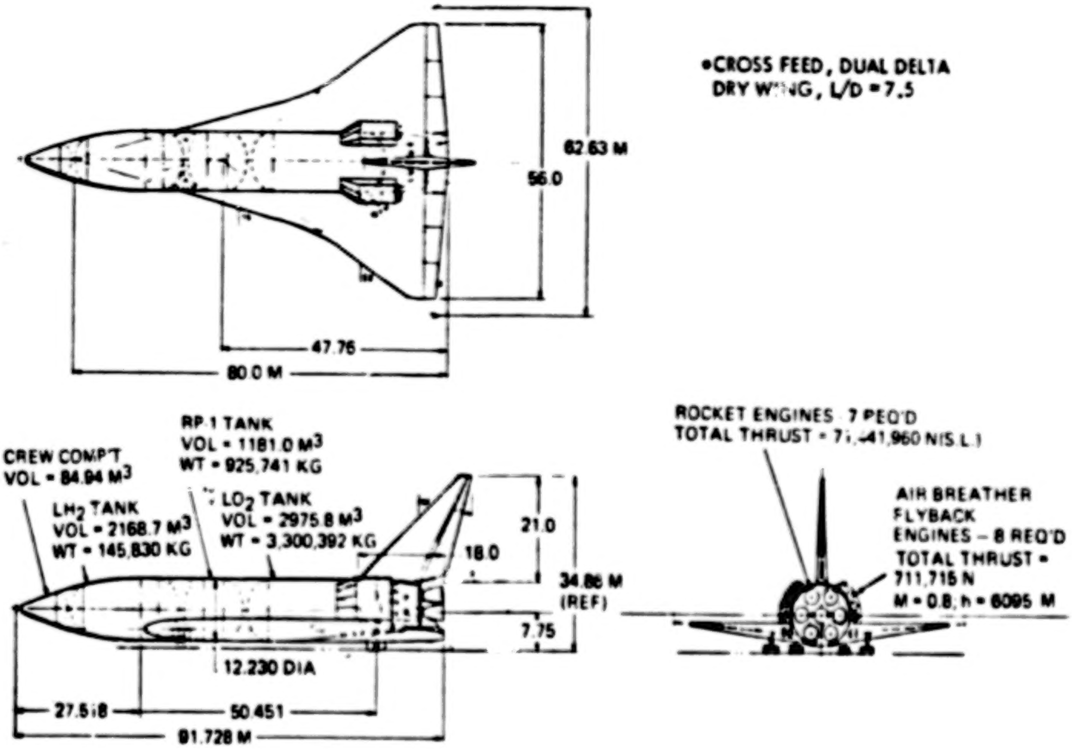


Figure 4.1-2. HLLV First Stage (Booster)—Landing Configuration

dry weight of 450,000 kg. Seven high P_c gas generator driven LOX/RP engines are mounted in the aft fuselage with a nominal sea-level thrust of 10 million newtons each. Eight turbojet engines are mounted on the upper portion of the aft fuselage with a nominal thrust of 89,000 newtons each. A detailed weight statement is given in Table 4.1-2. The vehicle propellant weight summary is projected in Table 4.1-3.

The HLLV orbiter is depicted in Figure 4.1-3. The vehicle is approximately 97 m in length with the same wing span, vertical height, and nominal body diameter as the booster. The orbiter employs four high P_c staged combination LOX/LH_2 rocket engines with a nominal sea-level thrust of 5.3 million newtons each.

The cargo bay is located in the mid-fuselage in a manner similar to the STS orbiter and has a length of approximately 28 m. The detailed weight statement and a propellant summary for the orbiter are included in Tables 4.1-2 and 4.1-3, respectively.

The vehicle can deliver a payload of approximately 231,000 kg to an orbital altitude of 487 km at an inclination of 31.6°. The vehicle relative staging velocity is 2127 m/sec (6987 ft/sec) at an altitude of 55.15 km (181,000 ft) and a first-stage burnout range of 88.7 km (48.5 nmi). The first-stage flyback range is 387 km (211.8 nmi). All engine throttling to limit maximum dynamic

Table 4.1-2. HLLV Weight Statement, kg×10^{-3} (lb×10^{-3})

SUBSYSTEM	2ND STAGE	1ST STAGE
FUSELAGE	103.41 (227.98)	130.73 (288.22)
WING	39.20 (86.41)	78.17 (172.34)
VERTICAL TAIL	5.70 (12.57)	7.21 (15.89)
CANARD	1.39 (3.07)	2.21 (4.87)
TPS	52.59 (115.94)	-
CREW COMPARTMENT	12.70 (28.00)	**
AVIONICS	3.86 (8.50)	3.40 (7.50)
PERSONNEL	1.36 (3.00)	**
ENVIRONMENTAL	2.59 (5.70)	**
PRIME POWER	5.44 (12.00)	**
HYDRAULIC SYSTEM	3.86 (8.50)	**
ASCENT ENGINES	26.93 (59.38)	67.45 (148.70)
RCS SYSTEM	9.59 (21.15)	**
LANDING GEARS	18.38 (40.51)	**
PROPULSION SYSTEMS	*	44.99 (99.18)
ATTACH AND SEPARATION	-	4.59 (10.12)
APU	-	0.91 (2.00)
FLYBACK ENGINES	-	28.55 (62.95)
FLYBACK PROPULSION SYSTEM	-	18.39 (40.54)
SUBSYSTEMS	-	25.76 (56.80)
DRY WEIGHT	286.99 (632.71)	(909.12)
GROWTH MARGIN (15%)	43.05 (94.91)	(136.37)
TOTAL INERT WT.	330.04 (727.62)	(1045.49)

*INCLUDED IN FUSELAGE WEIGHT
**ITEMS INCLUDED IN SUBSYSTEMS

Table 4.1-3. HLLV Propellant Weight Summary (×10^{-6})

	FIRST STAGE		SECOND STAGE	
	LB	KG	LB	KG
USABLE	9.607	4.358	3.481	1.579
CROSSFEED	1.612	0.732	(1.612)	(0.731)
TOTAL BURNED	7.995	3.626	5.093	2.310
RESIDUALS	0.040	0.018	0.020	0.009
RESERVES	0.045	0.020	0.024	0.011
RCS	0.010	0.005	0.018	0.008
ON-ORBIT	-	-	0.095	0.043
BOIL-OFF	-	-	0.010	0.005
FLY-BACK	0.187	0.085	-	-
TOTAL LOADED	9.889	4.486	3.648	1.655

pressure during the parallel burn mode is accomplished with the first or booster stage engines only (i.e., second-stage engines operate at 100% rated thrust during boost).

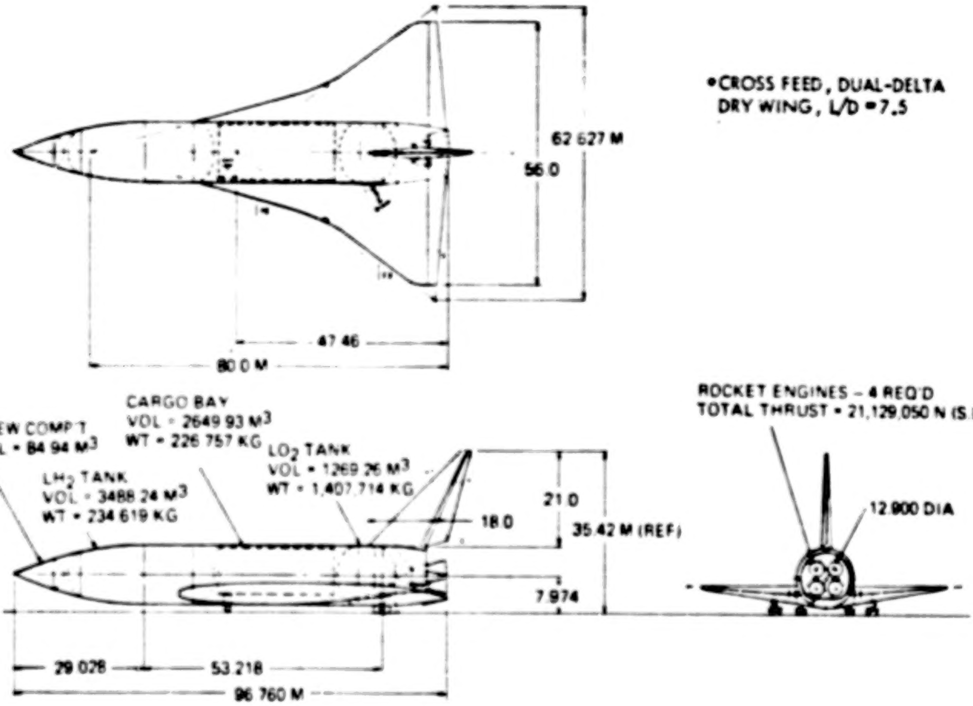


Figure 4.1-3. HLLV Second Stage (Orbiter) —Landing Configuration

4.2 SMALL HLLV CONCEPT (114K-kg payload)

The primary driver in establishing HLLV requirements is the timely delivery of construction material to LEO; thus, the payload magnitude becomes a major design parameter. The present day use of the term "heavy lift" connotes a launch system with a payload capability substantially greater than the 30 metric tons of the Space Shuttle. A "small" heavy-lift system is a large vehicle; the term "small" is comparative to the very large SPS reference system. While reduced HLLV size would permit use of the already developed SSME with appropriate modifications to provide longer life, this in turn incurs an increased number of flights to deliver an equivalent mass to orbit. In addition, VTO/HL vehicle size may be severely limited by erection, mating, and launch wind conditions. A final resolution of the most practical payload from overall considerations will have to await the results of separate future studies.

4.2.1 MATED VEHICLE CHARACTERISTICS

An alternate (smaller payload) configuration of more conservative design (i.e., more closely resembling the STS configuration) is depicted in the launch configuration, Figure 4.2-1. This configuration was adopted to permit the use of documented STS aerodynamic and performance data in order to address certain specific technical issues relative to VTO/HL vehicle concepts.

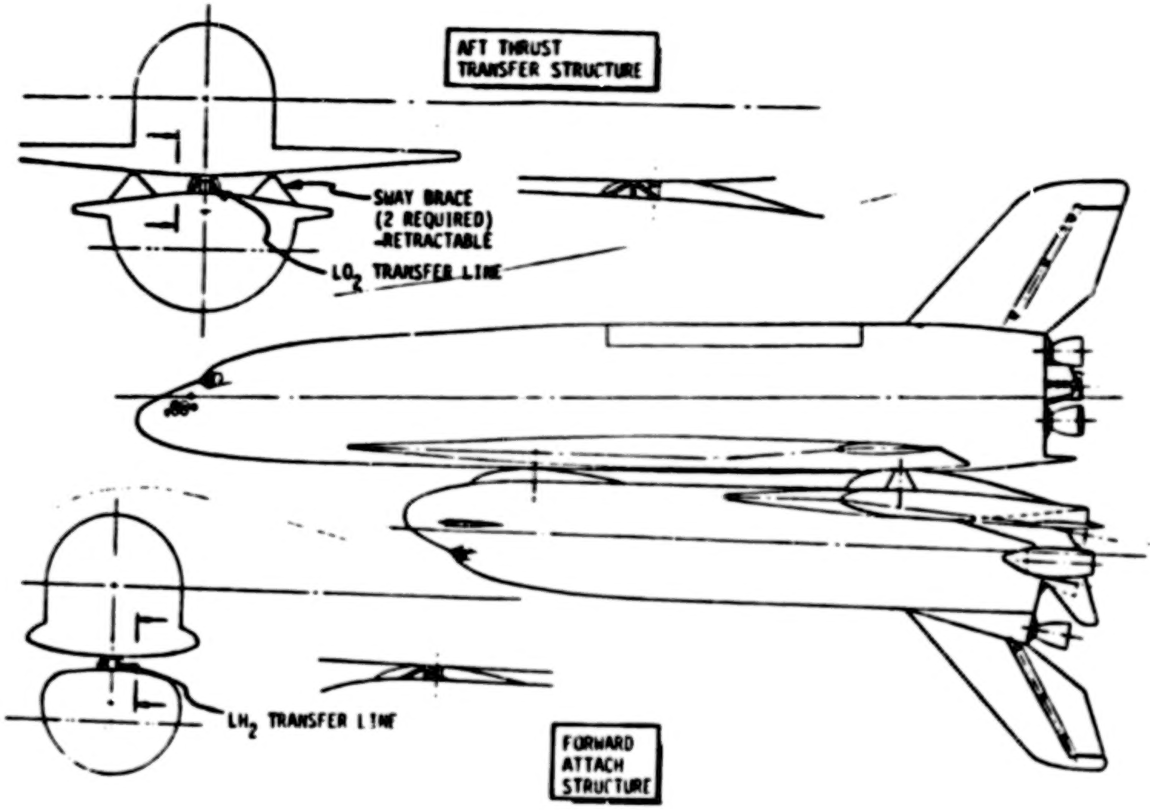


Figure 4.2-1. Mated System and Attach Structure

Each of the two stages has return-to-base capability with vertical takeoff and horizontal landing characteristics; the orbiter is unpowered at landing while the boosters fly back to the launch site with an airbreathing engine propulsion system. The launch vehicle utilizes a parallel burn propulsion mode with first-stage LO_2 and LH_2 being crossfed from the booster to the orbiter such that the orbiter stages with full propellant tanks. The booster utilizes high chamber pressure gas generator cycle LO_2/RP-1 fueled engines and the orbiter utilizes staged combustion LO_2/LH_2 engines developed from the Space Shuttle main engine (SSME) operating at zero NPSH.

The staging velocity was selected from earlier trade studies to be compatible with a heat sink structural concept for the booster. Material selection and development consistent with the 1990 time frame will ultimately play a significant role in the final selection of staging velocity. Thrust-to-weight requirements are selected to minimize engine size and crew/passenger discomfort. Orbital parameters are consistent with SPS LEO base requirements.

The mated system employs a fore and aft primary structural attach and sway brace attachment for differential roll stabilization. All attach points are released at staging through the application of explosive bolts.

The booster stage is approximately 61 m long and the orbiter, or second stage, is approximately 91 m long. Although the internal volume requirements are nearly the same, the boost vehicle employs eight LO_2/RP engines and, therefore, requires a wider base area. This wider base permits the application

of the "double-bubble" type propellant tanks to accommodate hypersonic aerodynamic stability requirements—hence, a foreshortening of the entire vehicle.

All ascent fuel to staging is contained in the boost vehicle. This necessitates a propellant transfer system. The LO_2 transfer system is supported by the aft structural attach system and is housed within the streamline fairing associated with the aft attach location. LH_2 is transferred at the forward attach structure. It is housed within the forward streamlined fairing. The streamline fairings are applied at drag and interference heating points.

The combined mass properties of the vehicle are presented in Table 4.2-1. At lift-off the HLLV weighs 3.56 million kg. At sea level the thrust of six orbiter engines is 10 million newtons, and the thrust of the eight booster engines is 35.6 million newtons. The total thrust at lift-off is 45.6 million newtons for a thrust-to-weight of 1.306.

Table 4.2-1. Combined Mass Properties

Condition	WT (10^6 kg)	X_o
Booster @ liftoff	2.410	2175
Booster @ liftoff	1.150	2262
Liftoff	3.561	2203
Booster propellant	-1.702	2127
Crossfed orbiter propellant	-0.446	
Staging	1.413	2320
Booster @ staging	-0.262	2573
Solo orbiter	1.151	2262
Orbiter propellant	-0.830	2367
Orbiter @ burnout	0.321	1991
Inert orbiter	-0.208	2015
Delivered payload	0.114	1950

During the booster flight of almost 160 seconds, 1.7 million kg of LO_2/RP are burned by the booster engines and almost 500 thousand kg of LO_2/LH_2 are transferred to the orbiter for SSME engine use. After separation from the booster at a relative velocity of about 1980 mps, the orbiter continues to orbit with a payload of 114,000 kg.

4.2.2 HLLV FIRST STAGE (BOOSTER)

The booster, Figure 4.2-2, employs hot structure with metallic heat sink, as required, for the entry flight regime of the booster. Initial investigations indicate that utilization of advanced metal matrix technology, wherever feasible, will result in a substantial weight savings.

The wing is sized to produce a nominal 333 km/hr landing speed and is optimized to minimize flyback propulsion requirements. Six turbojet engines are provided to accommodate the return-to-base mode after a launch. This flyback propulsion system weighs approximately 45,000 kg (with 9,000 kg of JP-5 fuel). Ascent propulsion is provided by eight advanced development engines of 4.5×10^6 newtons thrust each.

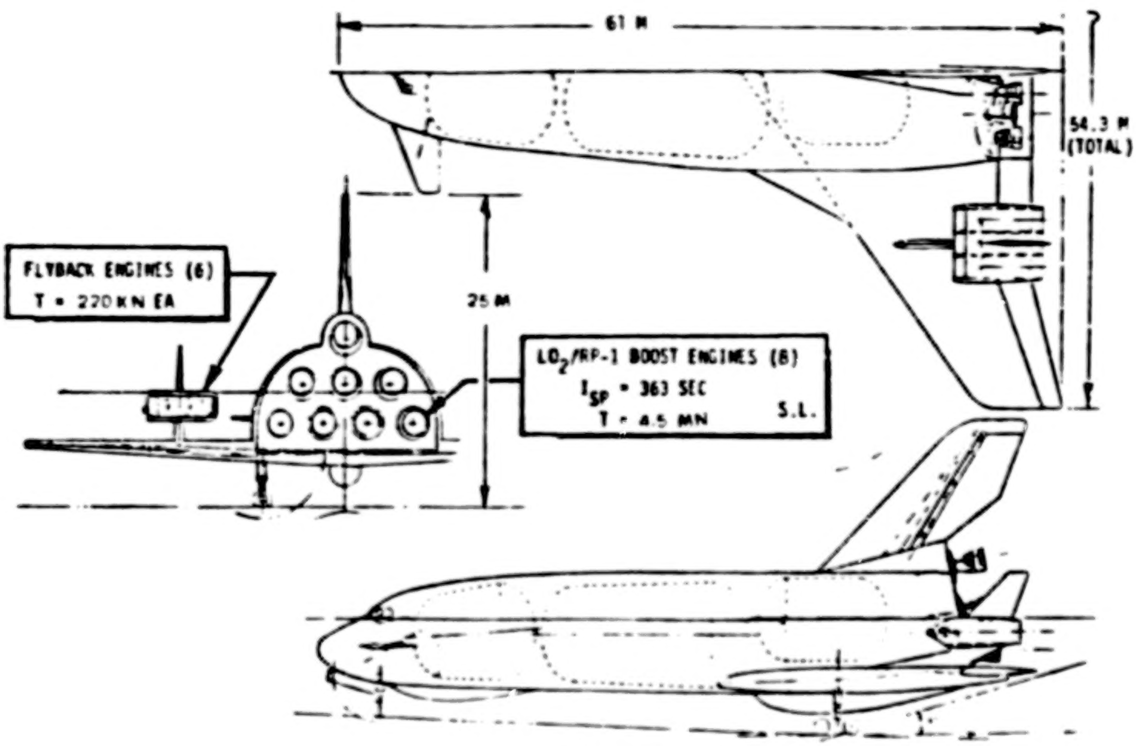


Figure 4.2-2. Satellite Power System—Booster

The system employs a belly-to-belly mating system for structural and propellant transfer continuity. Drag loads are reacted through a centerline attach truss located within the aft-mounted fairing, which also houses the LO_2 transfer line. The forward attach reacts yaw and pitch inputs and supports the LH_2 transfer line within the forward fairing. Retractable outboard sway braces (two) are employed to stabilize the system in differential roll.

The booster mass properties are given in Table 4.2-2. The structure represents about 37% of the dry weight. Of this total, 58% is fuselage, 32% is wing, 6% is tail, and 1.5% is canard. Use of advanced hot structure results in unit weights of 4.8 psf for the body surface area; 11.7, 8.5, and 8.0 psf for the planform area of the wing, tail, and canard, respectively. Allowances for a pressurized crew module for a crew of two have been provided. The landing gear weight was at 3.4% of the landing weight or 4.0% of the dry weight.

The propulsion system is almost 34% of the dry weight. Of this total, 51% is for engines, 18% for the RP tank, the orbiter crossfeed LH_2 and the combination LO_2 tank, 20% for the delivery systems, including the LO_2/RP feed and LO_2/LH_2 crossfeed systems, and 11% for the primary thrust structure.

A small auxiliary propulsion system for attitude control is provided. The flyback system represents 15% of the dry weight and includes feed and wet wing tankage for the propellant.

The total inert weight of the booster is also the staging weight and represents about 11% of the gross weight for a stage mass fraction of 0.89.

Table 4.2-2. Booster Mass Properties

ITEM	WT (kg·10³)	x_B
STRUCTURE	85.98	
TCS & PV&D	1.77	
LANDING GEAR	9.21	
PRIMARY PROPULSION	78.79	
AUXILIARY PROPULSION	1.13	
FLYBACK PROPULSION	34.47	
HYDRAULICS AND ACTUATION	8.05	
ELECTRICAL POWER	1.95	
AVAIONICS & EPD&C	7.17	
ECLSS	1.77	
PERSONAL PROVISIONS	0.81	
ORBITER/BOOSTER ATTACH STRUCT	1.00	
DRY WEIGHT	232.10	
RESIDUALS	3.66	
RESERVES	0.09	
LANDED WEIGHT	235.85	1671
USED IN FLIGHT	15.81	-
AUXILIARY PROPELLANT	0.91	-
FLYBACK PROPELLANT	9.07	-
STAGING WEIGHT	261.63	1683
BOOSTER—LO_2/RP	702.28	-
ORBITER—LO_2/LH	446.34	-
GROSS LIFTOFF WEIGHT	2410.26	1285

	x_B	$\%L_B$
LANDED	1671	73.3[illegible]
SEPARATION	1683	73.8[illegible]
GLOW	1285	56.4[illegible]

The booster lands with a c.g. of about 73.3% of the reference body length (L_B). At lift-off the booster has a weight of slightly over 2.4 million kg at a c.g. of 56.4% L_B.

4.2.3 HLLV SECOND STAGE (ORBITER)

The orbiter configuration, Figure 4.2-3, has been established to accommodate a payload of 114,000 kg in a volume of 1382 m³, with a payload bay length of 21.3 m. The payload density is 82 kg/m³.

The orbiter wing has been scaled from the Shuttle orbiter which permits the application of documented Shuttle orbiter aerodynamic data for performance estimation. The wing has been sized for the abort-once-around flight condition (payload onboard) to provide a nominal landing speed of 333 km/hr.

For the purposes of the present study, graphite-polyimide (GR/PI) has been selected as the primary structural material with RFCI tile for the TPS. Reentry thermal gradients are very similar to Shuttle orbiter because of the similar wing loading and planform. Thus, the RFCI can be tailored to accommodate the 600°F backface temperature allowable through the application of the GR/PI. It is assumed, for the time frame of the application, that a direct bond system will have been developed through the application of GR/PI. The structural weight fraction of the system is reduced by approximately 20% from conventional metallic structures.

The propulsion system employs six SSME engines which produce 2.1 NM thrust each (vacuum). The cryogenic tankage is non-integral to minimize the

requirement for a high-risk developmental technology. However, additional weight savings could be realized through the application of integral cryogenic tankage, but would require an intense design and development program to achieve the reliability, inspectability, and maintainability required for a reusable system.

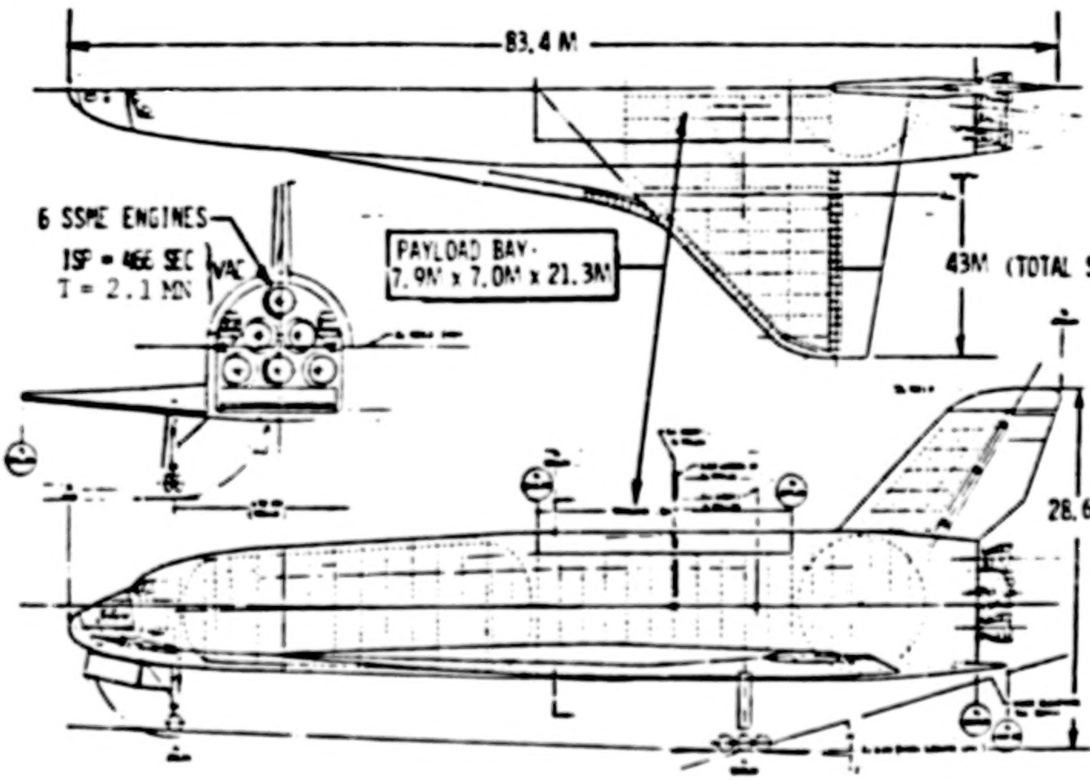


Figure 4.2-3. Satellite Power System HLLV—Orbiter

Additional weight savings have been realized by the judicious location of the avionics and ancillary systems. Communications between systems will be accomplished by the application of fiber-optics. Power supply systems will be located at the point of application (i.e., separate systems fore and aft), thus reducing the amount and run-length of the power cables.

The substantial increase in orbiter size, when designed for transporting much heavier payloads than the present Space Shuttle orbiter (29,500 kg), is readily apparent when the SPS HLLV orbiter is compared to the Shuttle orbiter at the same scale; see Figure 4.2-4. Dimensionally, such a comparison is somewhat misleading since the larger orbiter is a "wet" design, containing its own fuel, while the smaller is "dry."

The orbiter mass properties are presented in Table 4.2-3. The structure, when combined with the thermal protection system (TPS), represents almost 60% of the dry weight. Of this total, 66% is fuselage, 29% is wing, and 5% is tail. Use of advanced composite structure and reusable surface insulation results in unit weights of 5.9 psf for the body surface area, and 12.65 psf and 9.2 psf for the planform area of the wing and tail, respectively. Allowances for a pressurized crew module, for internal thermal control (TCS) and purge, vent, and drain (PV&D) have been provided. Landing gear weight was estimated at 3.4% of the abort weight, or 5.5% of the dry weights.

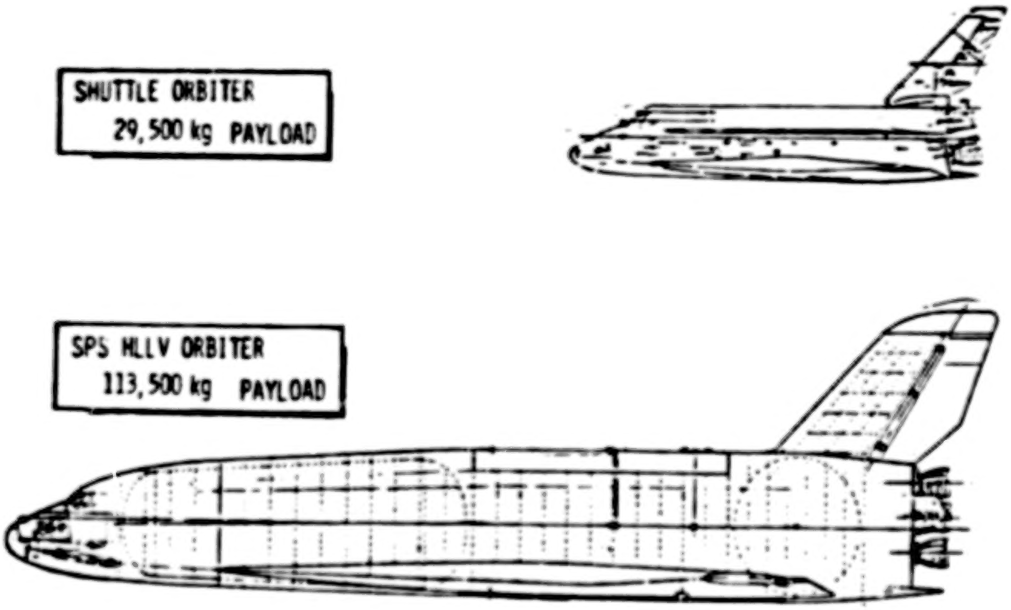


Figure 4.2-4. Size Comparison—Orbiters

Table 4.2-3. Orbiter Mass Properties

ITEM	WT ($kg \cdot 10^3$)	X_0
STRUCTURE	78.81	
TPS, TCS & PV&D	40.00	
LANDING GEAR	10.87	
PRIMARY PROPULSION	46.67	
AUXILIARY PROPULSION	2.06	
HYDRAULICS & ACTUATION	4.01	
ELECTRICAL POWER	1.95	
AVIONICS & EPD&C	7.68	
ECLSS	1.77	
PERSONAL PROVISIONS	0.81	
PAYLOAD PROVISIONS	1.13	
ORBITER/BOOSTER ATTACH STRUCT	1.00	
DRY WEIGHT	196.81	
RESIDUALS	0.55	
RESERVES	0.03	
LANDED WEIGHT	19[illegible] 03	1999
USED IN FLIGHT	6.36	-
AUXILIARY PROPUL. PROP.	3.14	-
TOTAL INERT WEIGHT	207.53	2015
PAYLOAD	113.5	1950
ABORT WEIGHT	320.93	1991
ASC PROPELLANT	830.14	-
GROSS LIFTOFF WEIGHT	1151.08	2262

	X_0	L_B	MAC
ABORT	1991	64.2%	13.9%
LANDED	1999	64.5%	14.8%
INERT	2015	65.0%	16.7%
GLOW	2276	73.4%	47.1%

The propulsion system is almost 24% of the dry weight. Of this total, 52% is for six modified SSME engines, 24% is for non-integral LO_2 and LH_2 tanks, 18% for delivery systems, including tank, crossfeed, fill, vent and drain lines, and valves. The basic thrust structure is 6.4% of the propulsion system weight.

The remaining systems weigh about 20,400 kg, or 10.5% of the dry weight. All weights are based on similar elements of the STS orbiter. The auxiliary propulsion system (APS) is basically that of the STS orbiter, while the hydraulic system is double that of the STS orbiter. Two redundant/separate fuel cell/cryo tank sets are employed—one for the forward equipment, and the other for the aft equipment. Two redundant and separate environmental control systems are also provided. The forward system also includes the life support system. The avionics are located functionally and are connected only by fiber optical wiring.

Personnel provisions are for a crew of two for two days. Allowances are provided for payload installation and mechanical/electrical/fluid connections to the booster. Residuals account for trapped line and tank fluids and gases. The reserves are for the APS. Almost 9,500 kg of fluids are used during ascent, flight, and descent including 3130 kg of APS propellants.

The total inert weight represents about 18% of the gross weight, and the payload 10%, for an overall stage mass fraction of 0.72.

The orbiter normally lands with a center of gravity (c.g.) at 64.5% of the reference body length (L_B = 7900 cm) or 14.8% of the mean aerodynamic cord (MAC). The abort c.g. is only slightly aft of the normal landing c.g. From ground lift-off to booster separation, the orbiter weight is slightly greater than 1.13 million kg with a c.g. at 73.4% L_B or 47.1% MAC.

4.2.4 VEHICLE TRAJECTORY DATA

Ascent constraints used for the SPS HLLV ascent simulation included 3-g maximum sensed acceleration and 3200 kg/m² (650 lb/ft²) maximum dynamic pressure. Only the booster engines were throttled in the mated configuration to meet the acceleration constraint, thereby reducing the engine gimbaling requirements. The vehicle was guided by a series of inertial pitch rates with the inertial frame located at the launch site at the moment of launch. The orbiter was targeted into a 95 by 318 km orbit inclined 31.1 degrees.

The vehicle ascent trajectory time history is depicted in Figure 4.2-5. Several aerodynamic maneuver schemes for booster entry and flyback were simulated while adhering to constraints of 3-g maximum sensed acceleration and 3200 kg/m² maximum dynamic pressure. Definition of the airbreathing propulsion system weight then enabled complete simulation of the launch vehicle ascent performance. The Program to Optimize Simulated Trajectories (POST) was used for the simulations. The recovery scheme selected is presented in Figure 4.2-6.

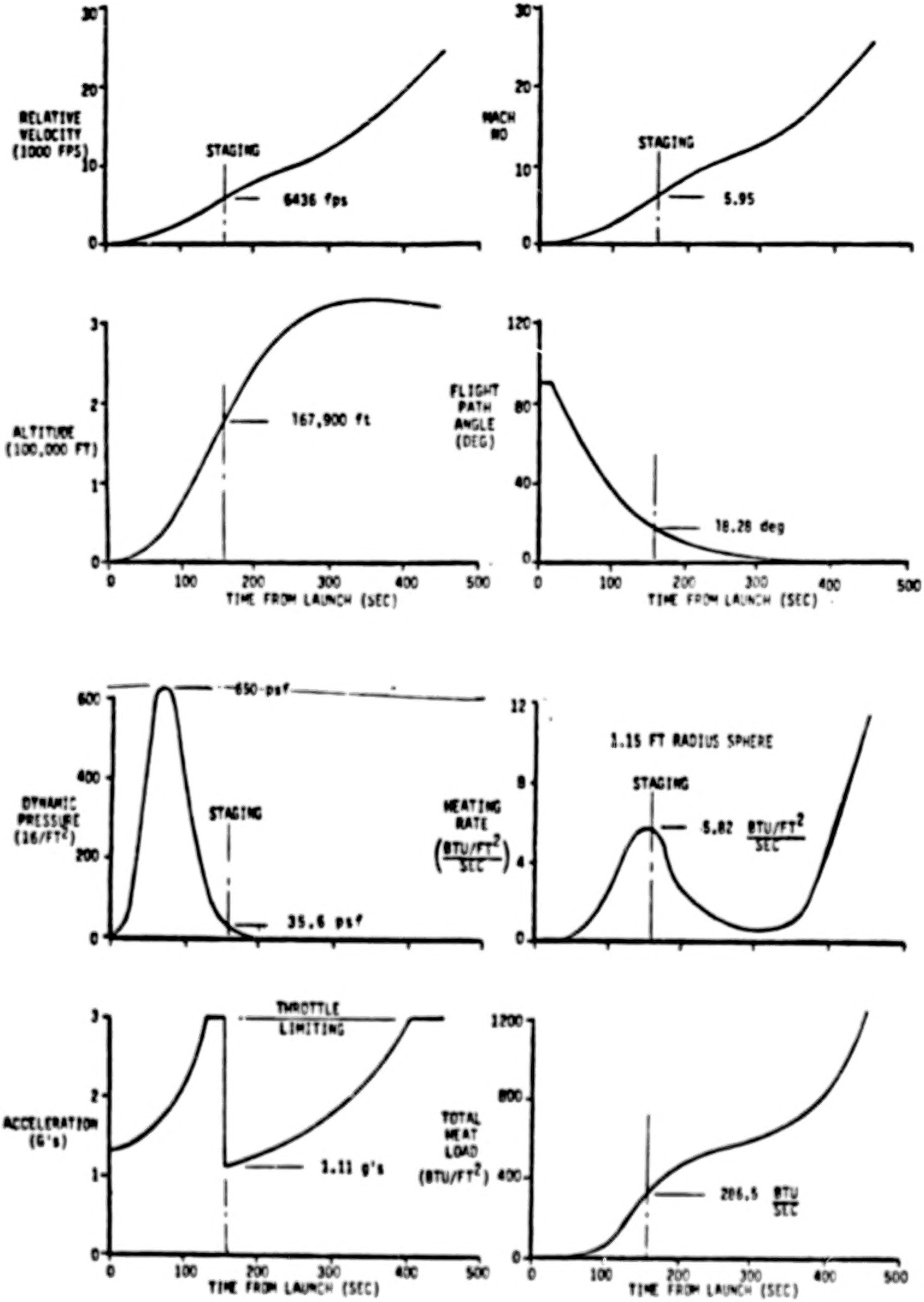


Figure 4.2-5. Ascent Trajectory Time History

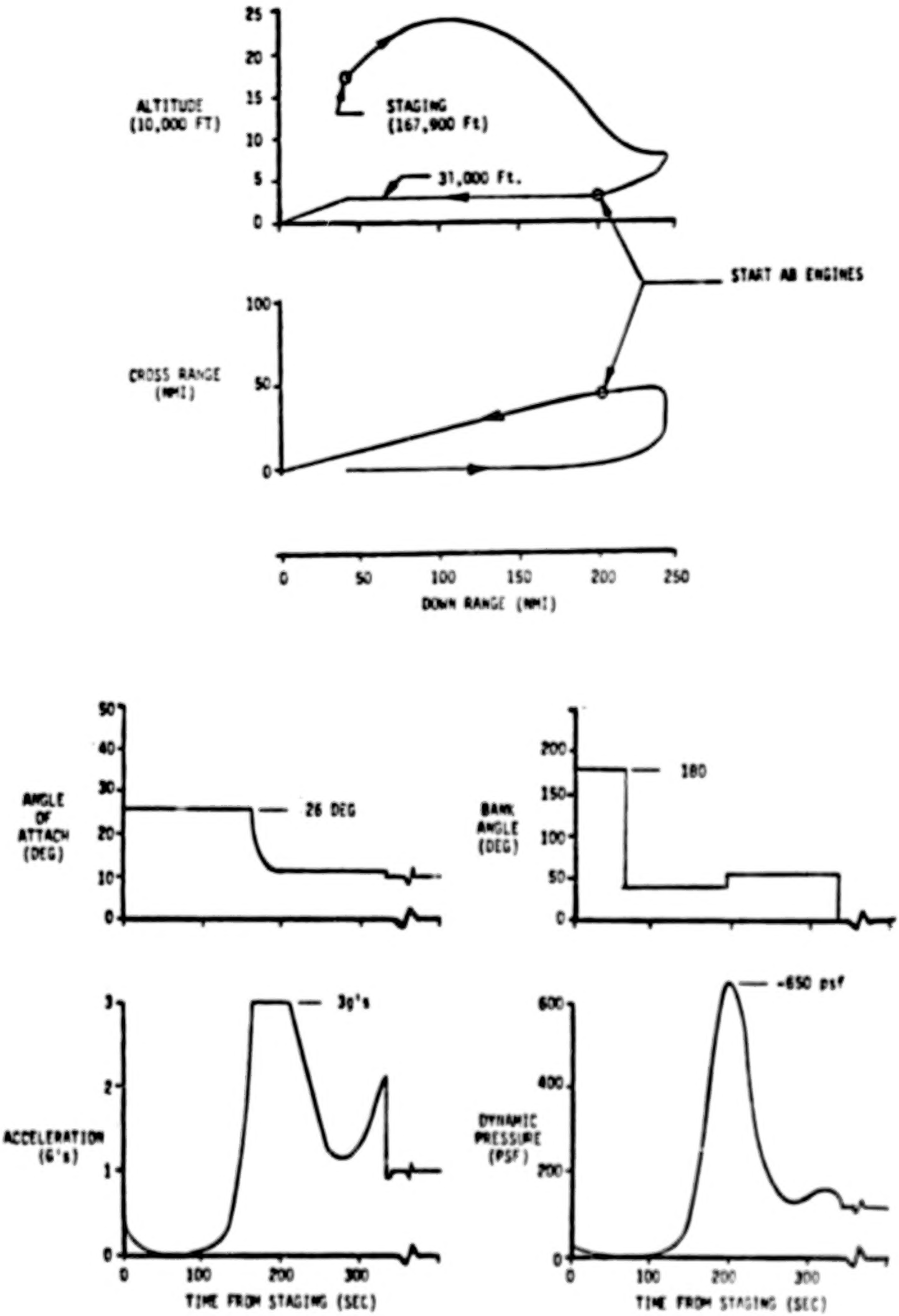


Figure 4.2-6. Booster Flyback, Out-Of-Plane Maneuvers

During the first 160 seconds after separation, the booster is guided through a dual-fixed bank maneuver which was shown in early Shuttle studies to provide significant cruiseback range reduction. From separation to apogee, the bank angle is held at 180°. After apogee, the booster is flown at the largest bank angle that does not cause the dynamic pressure limit to be exceeded.

When the 3-g limit is reached at approximately 160 seconds, the angle of attack is modulated to hold a 3-g pull-up. Angle-of-attack modulation is terminated at maximum dynamic pressure (195 sec).

From 195 seconds after separation until the booster is completely turned around, the bank angle is fixed at the value which maximizes booster energy at completion of the turn.

The booster glides at maximum L/D down to the cruiseback altitude of 9500 m. The 296 km cruise to the launch site proceeds at Mach 0.6. The cruise conditions were selected to minimize flyback propulsion system weight.

4.3 TECHNICAL ISSUES ASSESSMENT

The reference HLLV concept adopted to satisfy the Satellite Power System study requirements for earth-to-orbit transportation is a two-stage vertical launch, horizontal landing parallel burn configuration utilizing winged vehicles and booster/orbiter propellant crossfeed. Although a preliminary definition of vehicle design has been identified, several technical concerns have been identified which require further analyses to assure configuration suitability. These concerns include:

- Vehicle flight characteristics during entry and low speed
- Ascent control requirements
- Distribution of thrust loads and structural requirements
- Preliminary thermal/structural assessment

These issues have been addressed to the extent possible with available resources and are discussed below.

4.3.1 VEHICLE FLIGHT CHARACTERISTICS

The first of the four technical issues addressed during this study phase revolved about the central issue of vehicle flight characteristics. Both the booster and the orbiter are characterized by far aft centers of gravity incurred by the heavy mass of the propulsion systems. In addition, both vehicles must be aerodynamically stable and controllable throughout a wide range of flight speeds and attitudes. This is also characteristic of the Space Shuttle orbiter.

Over the many years that lifting entry spacecraft have been studied and configured, the aft center-of-gravity design challenge has induced perhaps as many configuration approaches as there have been designers. The Shuttle orbiter design accommodated the aft c.g. problem by stringent wing planform aerodynamic design and by careful mass distribution. The concentration of mass at the aft of the fuselage in the present concepts reopens the challenge.

Two approaches are represented in the present HLLV designs: adoption of the Shuttle orbiter planform for the present orbiter, since the c.g. ranges are essentially equivalent; and far aft, control-configured wing arrangement for the booster to balance an even farther aft c.g.

The final resolution of the configuration arrangement(s) will require extensive analyses and ground test programs.

The individual booster and orbiter aerodynamic characteristics have been estimated using established analytical techniques and by adjusting the Space Shuttle aerodynamic data. The analytical techniques include USAF DATCOM and the recently completed digital USAF DATCOM aerodynamic code.

The interference drag for the parallel mated ascent configuration was estimated by evaluating the U.S. Air Force Flight Dynamics Laboratory studies conducted for mated vehicles.

The orbiter stage essentially designed to exhibit the Space Shuttle wing loading and wing geometry. Therefore, the Shuttle drag, lift-curve slope, and lift-induced drag were utilized, with adjustments made to the drag to reflect changes in skin friction and base drag to account for orbiter fuselage differences from the Shuttle.

The booster stage aerodynamics were obtained primarily by analytical methods, the major of which was the USAF digital DATCOM aerodynamic code.

The mated ascent configuration lift-curve slope was assumed to be effectively equal to the orbiter-alone values since, due to the close proximity of the mated vehicle wing, the summation of both vehicle lift-curve slopes is unrealistic. Trajectory analysis indicated that, due to very small angles of attack during ascent, the lift-curve slope was only of minimal consequence to the trajectory. Therefore, the lift-curve slope of the largest wing configuration, namely the orbiter, was utilized.

The booster wing was sized to provide acceptable landing speed and has an aspect ratio of four. The wing offers a good subsonic lift-curve slope, low drag due to lift, and a relatively high subsonic lift-to-drag ratio for the booster configuration.

The orbiter wing selection was tailored to nearly match the Shuttle orbiter to allow the STS orbiter configuration to remain within the Space Shuttle thermal and loads environment. As a result, the aerodynamics for the orbiter reflect essentially those of the Shuttle.

Baseline aerodynamic characteristics are presented in Figure 4.3-1.

Ascent trajectories were simulated with drag estimates which include a 20% increase due to interference between the booster and orbiter vehicles. Mated ascent configuration zero-lift drag and lift-curve slope are shown in Figure 4.3-2.

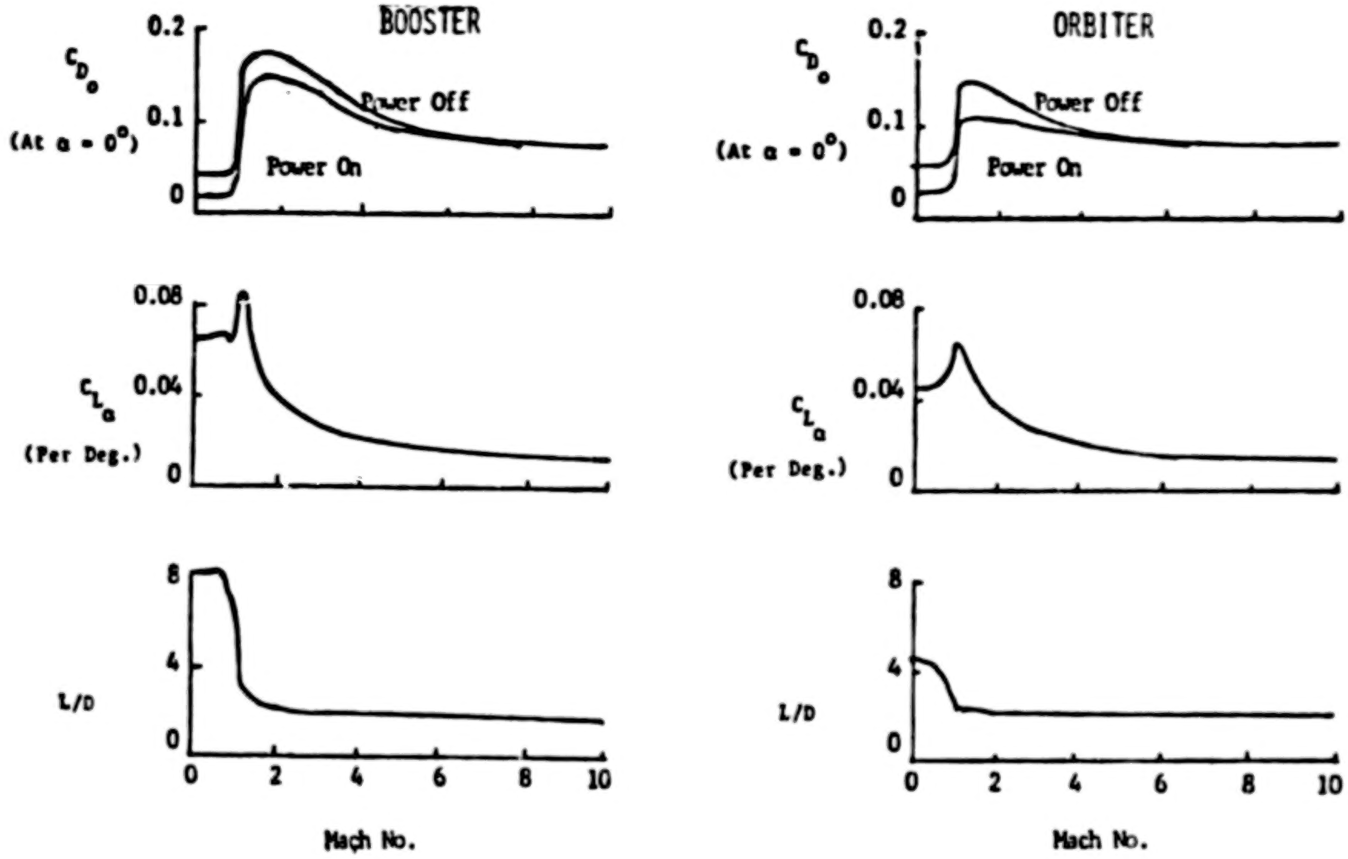
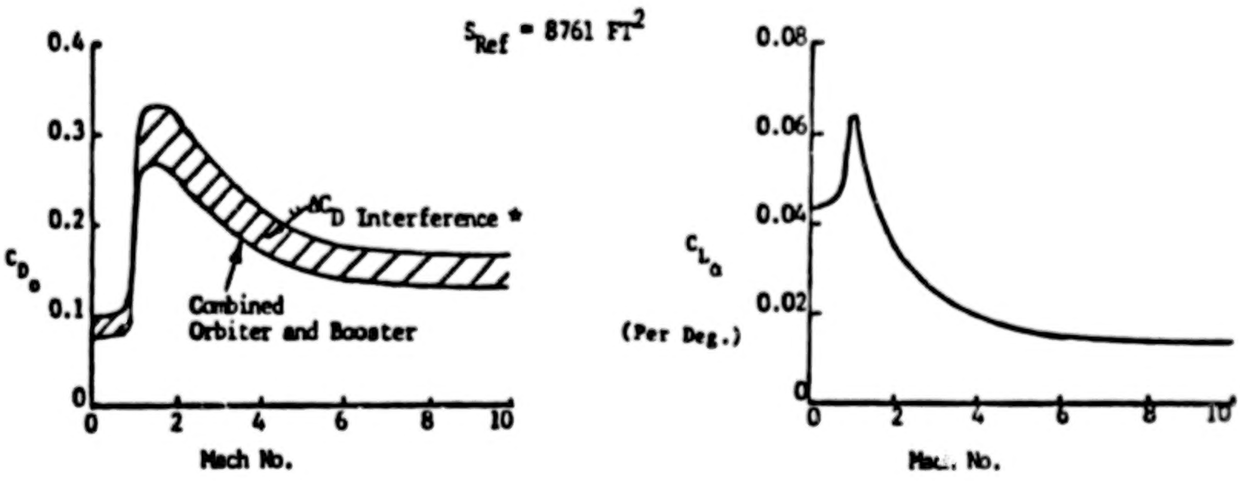


Figure 4.3-1. Baseline Aerodynamic Characteristics



* REF.: AFFDL-TR-70-21; ΔC_D due to interference can be as high as 20% above combined C_{D_o}

Figure 4.3-2. Mated Ascent Configuration Zero-Lift Drag and Lift-Curve Slope

Potential problems which have been identified in the area of vehicle flight characteristics reflect those which have been identified during earlier studies of similar launch and energy configuration concepts. Interstage interference is a very real concern in the present Space Shuttle program and is presently receiving close attention. The complex flow fields of the interference region create substantial acoustic and dynamic loads as well as aerodynamic drag penalties. Early Shuttle wind tunnel test data have shown that stage-to-stage gap minimization and fairings can provide significant reductions in adverse effects, but at some cost if incorporated as a program change; early treatment can be effective. The far aft c.g. characteristics of the lifting launch and entry vehicles, due to the very high propulsion system masses, require close attention to mass distribution and aerodynamic shaping.

Transportation of the orbiter from the point of manufacture or alternate landing sites also requires early attention. Airbreathing engines are not incorporated in the vehicle to save weight so the orbiter cannot operate in a ferry mode. Some form of an auxiliary propulsion system is necessary, since the development cost of a suitable carrier aircraft would very likely be prohibitively expensive. The design and operation of very large and outsize aircraft systems incur a new level of design analyses; such challenges have been met in the past as necessary in the cases of the B-29, the 747, and the C-5 aircraft and require recognition of the large masses, inertias, and dimensions involved.

Analytical studies of aircraft that are characterized by large inertias, less than optimum mass distributions, and just very large dimensions must receive early attention. The combination of these characteristics can provide a difficult design situation at later stages, and should be resolved or at least treated in general terms during initial development stages to help forestall later complications.

A significant amount of such effort has been accomplished during the early Space Shuttle and pre-Shuttle conceptual analysis programs as well as during development programs for conventional aircraft. These data are available in various public and private archives. Some effort should be expended in each technical area to extract this volume of background data in order to reduce the very real chance of "reinventing the wheel" again.

Once promising configuration concepts are identified, a comprehensive series of wind tunnel tests should be conducted to provide a solid foundation for further configuration development and design excursions. This is particularly true where the present configurations depart from those tested earlier, as noted above. In addition, design approaches to operation of large systems with very widely separated subsystems should be simulated to verify approaches to distributed functions with local control.

4.3.2 ASCENT CONTROL REQUIREMENTS

The concerns that tend to dominate this area involve main engine gimbal and throttle requirements, and the impact of aerodynamic flow interaction effects.

In the parallel mode of engine operation, the center of gravity of each vehicle is substantially offset from that of the mated launch vehicle. In addition, this mated c.g. moves substantially from an intermediate position at launch toward the orbiter which retains all of its fuel during first-stage ascent. The far aft location of the c.g. can also require large gimbal angles at the engines to "track" the c.g. as it moves. The proper orientation of the engine nominal (null) angles and selection of control philosophy can go a long way toward reducing thrust cosine losses during ascent and reducing thrust "kick" loads on the thrust structures.

Aerodynamic interaction effects occur because each vehicle operates within the flow field of each other. At supersonic speeds, the impinging shock waves are reflected back and forth between the vehicles and may ultimately result in a standing normal shock at some position depending on the relative proximity of the vehicles. This highly complex flow field alters the stability, control, and performance of the system in general as well as in local areas, generally to a degraded level.

The final resolution of these issues will require extensive evaluation of those problem areas which are identified as the configuration evolves during development.

During booster burn, the composite center of gravity shifts from a position in the booster to a position in the orbiter in a manner illustrated in Figure 4.3-3. The initial forward shift is due to the use of propellants in the top of the booster tanks with a composite c.g. forward the the combined vehicle c.g. As propellant is consumed, the composite propellant c.g.'s move aft as does the combined vehicle c.g.

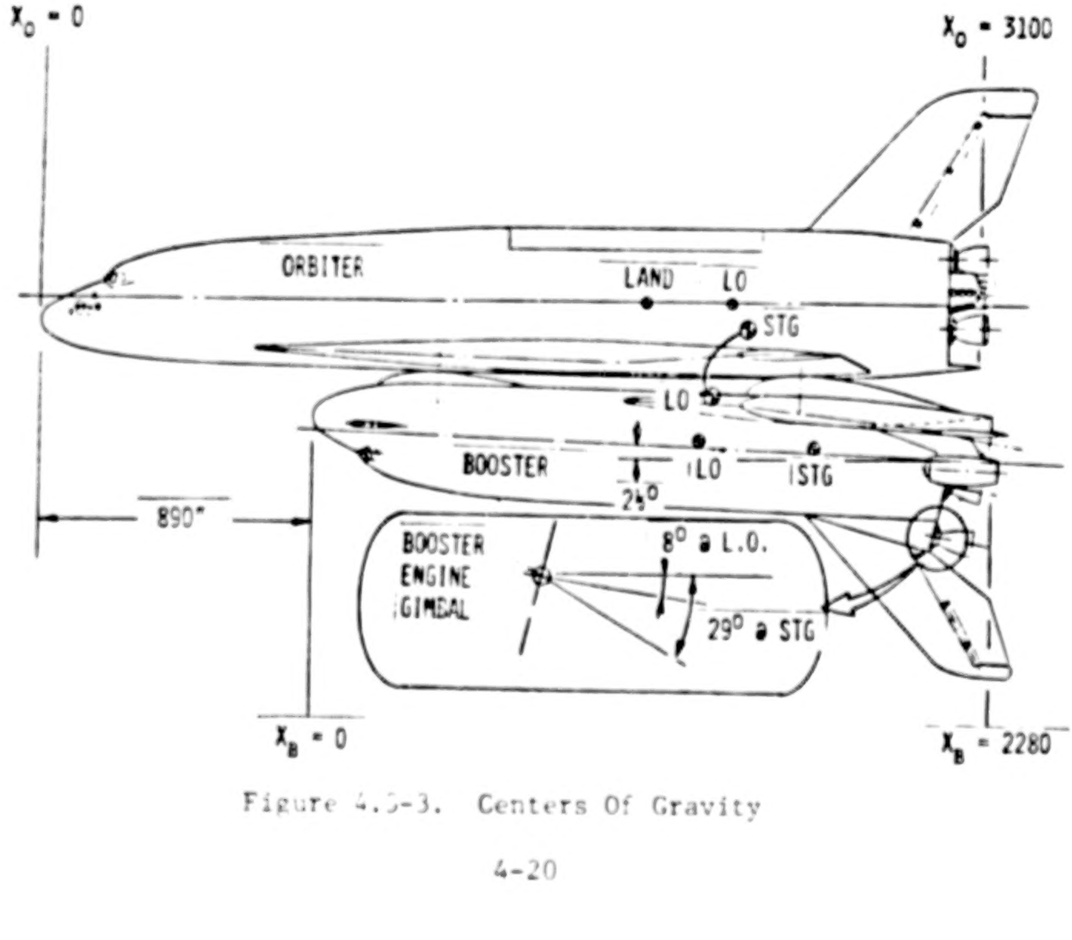


Figure 4.3-3. Centers Of Gravity

If the null position of the orbiter engines is along the orbiter centerline, the booster engines must be gimbaled to balance the thrust loads about the traveling c.g. Preliminary estimates indicate the booster engines must be deflected 8° from the orbiter centerline or 10.5° from the booster centerline at liftoff and 28° (or 30.5°) at booster burnout. The integrated time history results in a mean deflection of about 17°. The cosine vector for this deflection results in a 3.5% loss of total thrust.

After booster shutdown and prior to separation, the orbiter engines must be deflected about 7° for the thrust centerline to pass through the combined vehicle c.g. At separation, the engines must be returned to the normal position to align with the orbiter c.g.

Complex flow interaction between parallel vehicles will produce high local heating and pressure loading on the inner surfaces. In addition to local loads, the flow interaction may produce unsymmetric load distributions which would require trajectory steering commands. In addition, the same unsymmetric load distributions may have an influence on separation dynamics and may require additional attitude control during separation.

For atmospheric staging, the flow interference-caused pressure distribution has to be known in order to determine safe separation maneuvers and to select the optimum separation maneuver.

Bow shocks from the nose or leading edges of each vehicle intersect and interact with the surface boundary layer, Figure 4.3-4, causing local high pressures and boundary layer separation and reattachment. Typically, high heating rates are experienced at the boundary layer reattachment zone.

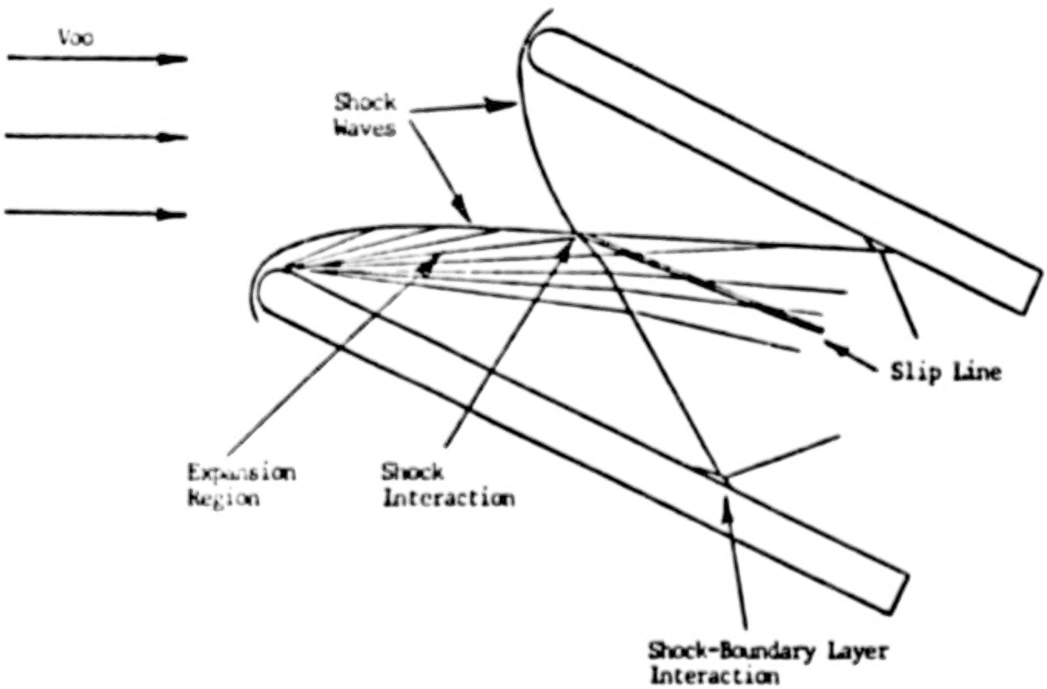


Figure 4.3-4. Typical Flow Interference between Parallel Surfaces

The flow interaction between parallel vehicles will result in incremental normal and axial loads as well as pitching and yawing moments on the separating stages, as shown in Figure 4.3-5. These increments will vary with the lateral and horizontal separation distances, and will influence the selection of separation maneuvers and/or attitude control requirements.

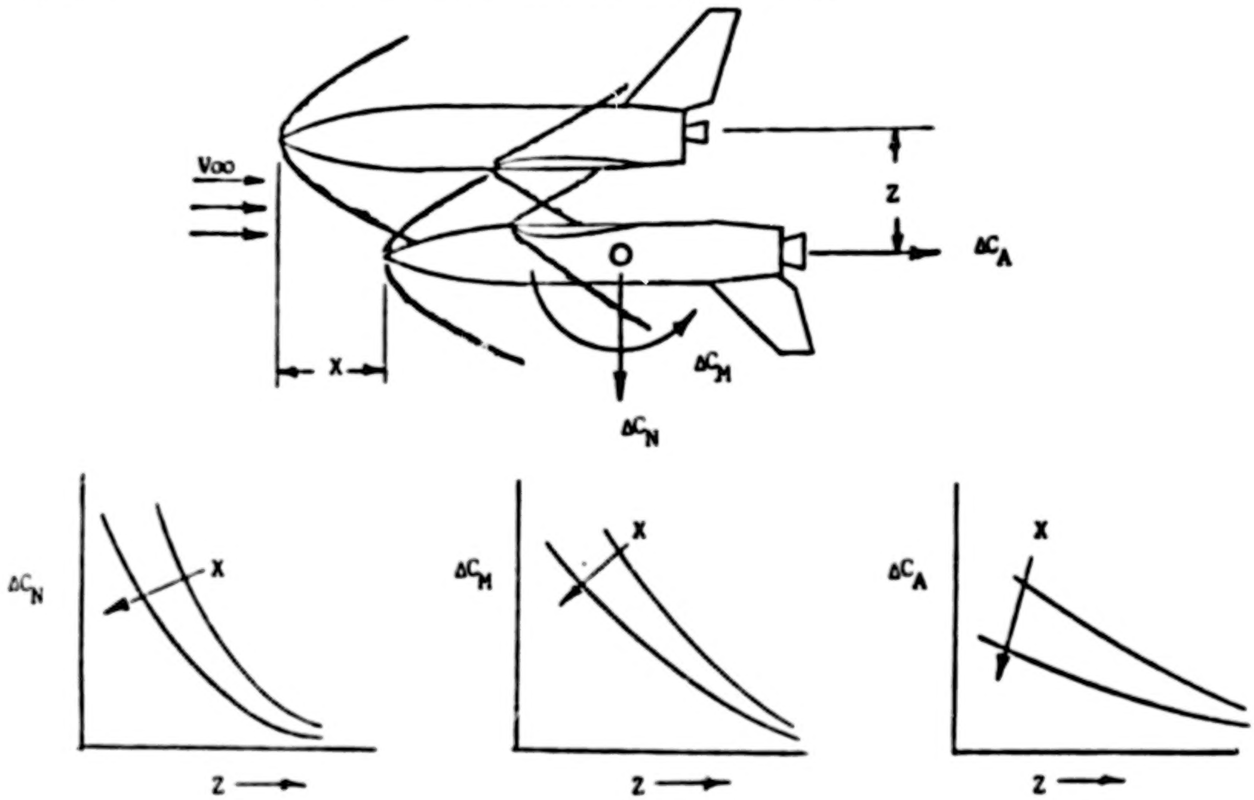


Figure 4.3-5. Separation Incremental Aerodynamics

The most significant problems which may be forecast relate to the interference flow field between the mated and separating vehicles. Interstage interference during mated flight operation can probably best be reduced by avoiding those design conditions which could induce interference. Shuttle orbiter wind tunnel tests have demonstrated the beneficial effects of reducing the length of the forward attach strut length to a safe minimum or by providing a booster-attached aerodynamic fairing. (These modifications were not adopted for the Shuttle system because they constituted significant and costly design changes at the time they were developed.)

The stage separation of the vehicles is a second area impacted by the interference flow field. Either adverse or proverse loads can be induced on the vehicles, depending upon their particular aerodynamic configurations. General analyses can be performed but, in general, the final determination can only be derived from the results of detailed wind tunnel tests. Here, again, the pre-Shuttle data archives could yield substantial background information on a wide variety of configuration concepts.

There appears little need for analysis of a generally applicable nature at this point. Most of the ascent control concerns for the SPS launch vehicles

are very nearly those considered in exhaustive depth during the development of the Shuttle system and during the earlier preliminary developmental studies. Until the configuration development has proceeded somewhat farther and has settled on a more specific concept, not too much new analysis is required. Some effort can be expended on trade studies using existing data on the relative advantages of engine gimbal versus aerodynamic control during boost and separation, and on the controlling role of center engines of a multiple engine arrangement such as that on the present booster.

Once candidate SPS HLLV launch system configurations have been defined in more detail, a series of wind tunnel tests to identify the effects of flow field interference would be particularly beneficial. Such tests would be of a "matrix" nature at first and, subsequently, of a computer-controlled trajectory nature.

4.3.3 THRUST LOAD DISTRIBUTION/STRUCTURAL REQUIREMENTS

The technical issue addressed is the concern regarding distribution of thrust loads between the two ascent stages and engine thrust structure concepts for each vehicle. The delivery of very heavy payloads to orbit infers the need for a large number of rocket engines on each vehicle (booster and orbiter); optimization studies almost invariably place the larger number by far on the booster in order to maximize the payload mass fraction of the orbiter. The distribution of distributed and combined thrust loads to the airframes requires stringent design emphasis.

The thrust loads are necessarily different and variable on each vehicle during ascent. These differences result in differential loads between the vehicles. The resolution of these forces drives the design of the interstage attach structure. The attach system must not only transmit these loads but must also resolve aerodynamic and dynamic loads while being easily and reliably separable during the staging maneuver. The attach structure is a natural location for locating the propellant crossfeed lines due to their inherent structural rigidity. This is implicit in the Shuttle orbiter design, in which the feed lines from the external tank interface with the orbiter in the aft attach structure assembly.

The ultimate resolution of these issues will require substantial design and analytical effort and detailed ground test verification of candidate design concepts.

Attach structure shear load is shown in Figure 4.3-6 as a total force in pounds versus trajectory time during ascent to staging point at t = 153.8 sec. At approximately t = 135 seconds, booster engines are throttled such that total vehicle acceleration is held to 3 g. Maximum shear load in the attach structure is 2.1 million kg just prior to staging and engine shutdown. The effect of booster engine throttling can be seen as a change in the slope of the attach load history.

The attach structure, Figure 4.3-7, consists of a fore and aft centerline truss system with two sway-brace attachments located in the lateral plane of the aft attachment. Both the truss attach structure and the propellant transfer

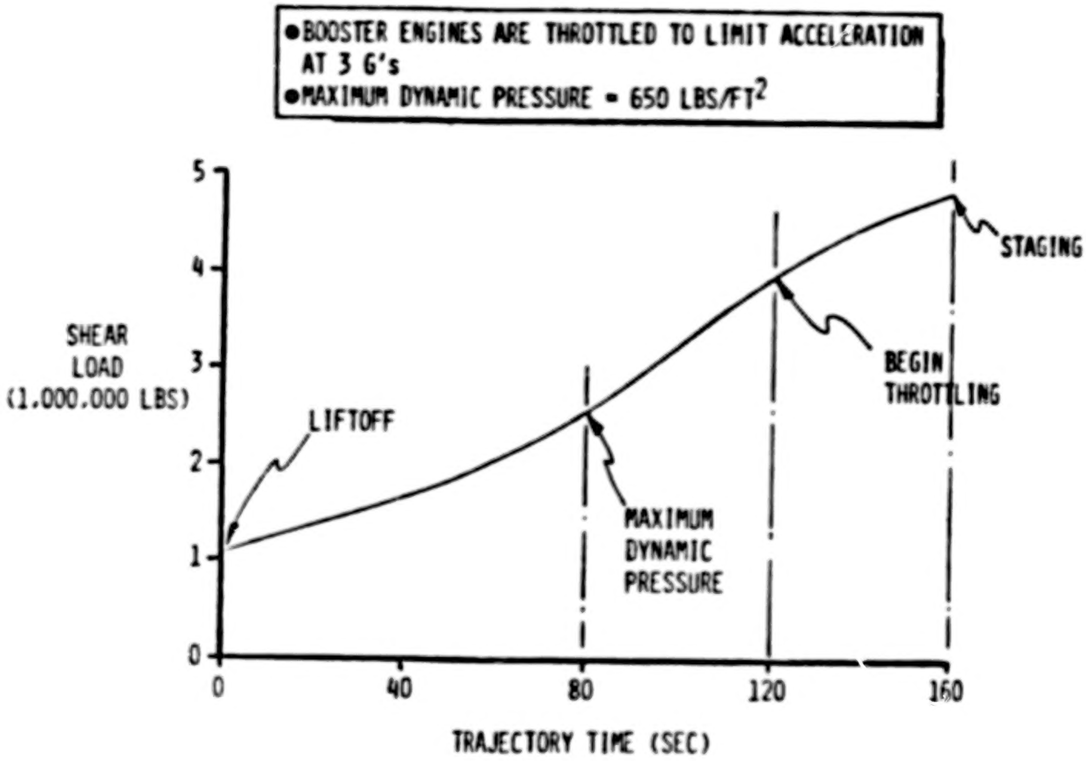


Figure 4.3-6. Attach Structure Shear Load

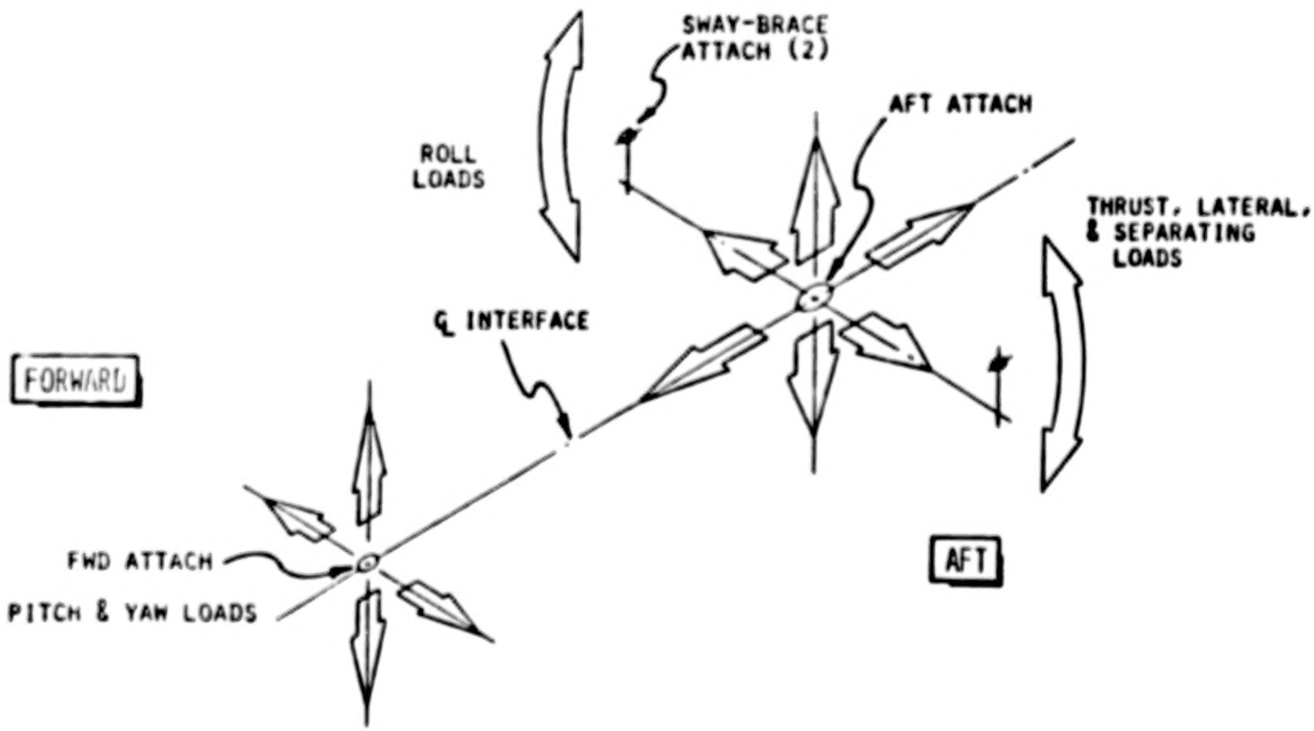


Figure 4.3-7. Attach Structure Schematic

lines (which are supported from each of the truss structures) are located in centerline streamline fairings on the booster stage.

The aft truss system is employed to resolve the primary loads—longitudinal thrust, drag, side loads, and vertical loads. The forward truss system is employed to resolve vertical and side loads, and asymmetric thrust loads. The sway braces are employed to stabilize the mated system in differential roll.

The orbiter thrust structure, Figure 4.3-8, is similar to that employed on the Shuttle orbiter. Both vehicles utilize the same engines; however, for the HLLV orbiter, six engines are used rather than three.

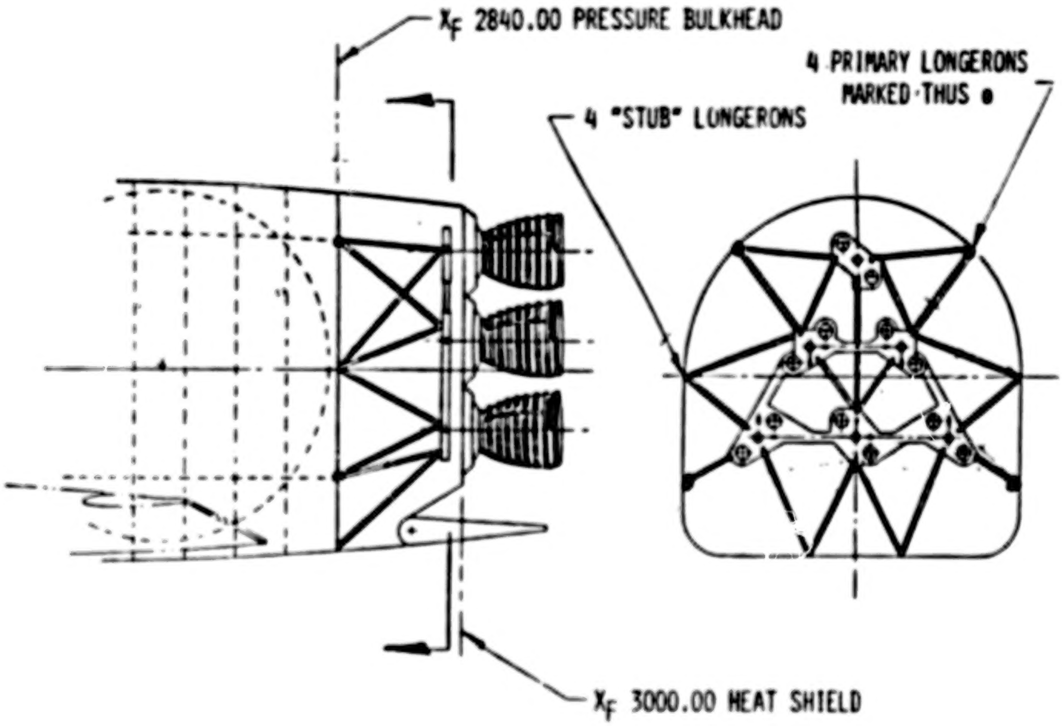


Figure 4.3-8. Thrust Structure—Orbiter

Engine arrangement is established to produce the maximum thrust vector as close to the lower mold line as possible. This minimizes the amount of engine gimbaling required during the ascent portion of the flight. Transverse beams are employed to support the two lower rows of engines, three and two each, with the sixth engine supported from an attached but isolated space structure. The transverse beams are supported from a built-up space structure similar to that employed on the Shuttle orbiter, with the termination of this structure occurring at the four main longerons and at four "stub" longerons located to accommodate the geometry of the thrust structure.

Material for this structure will be FRAT with specific shapes and joints created by the utilization of the SPFDB process. GR/PI will be applied as required to provide required stiffness with a minimum weight penalty.

Eight SSBE-type engines are required to provide ascent thrust for the boost vehicle. Two transverse beams are employed to support the two lower rows of engines, four and three each, with the eighth engine supported from an attached but isolated space structure, Figure 4.3-9. The transverse beams are supported from a built-up space structure similar to that employed on Shuttle orbiter with the termination occurring at the five main longerons and at four "stub" longerons located to accommodate the thrust structure geometry. Because of the reach from the center engines to the longeron attach points it may be necessary to provide a secondary structure for continuity. It appears feasible to employ a secondary bulkhead to provide a potentially less complex method for thrust distribution. The secondary bulkhead could also be utilized for equipment mounting.

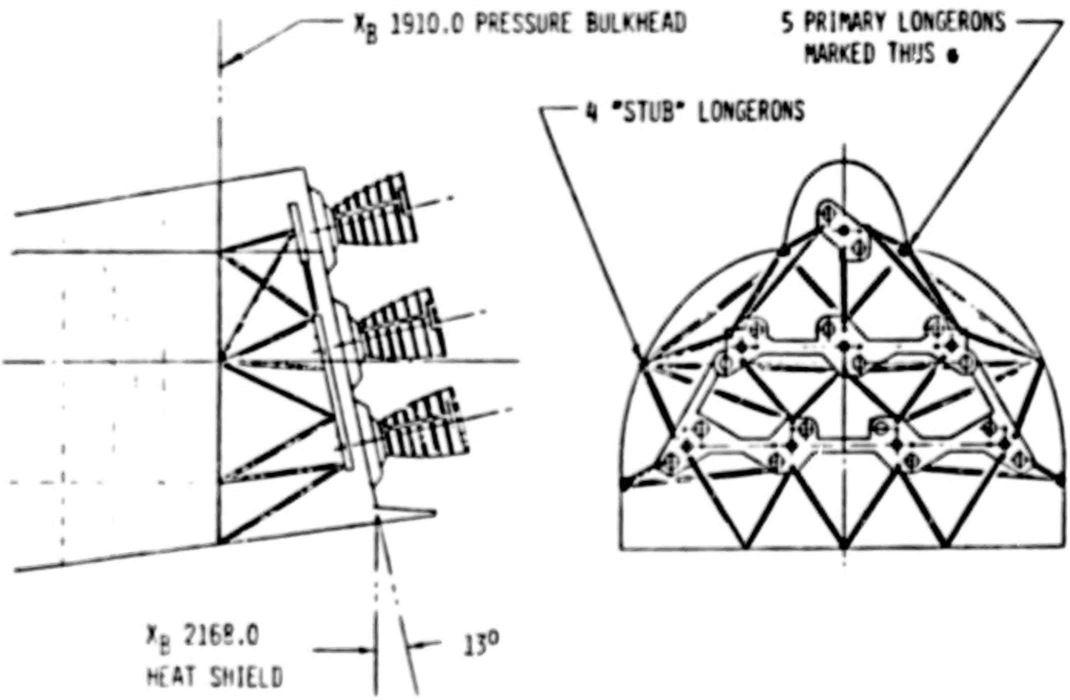


Figure 4.3-9. Thrust Structure—Booster

Material for this structure will be FRAT with specific shapes and joints fabricated by application of the SPFDB process. Where required for stiffness, composite materials will be applied to the FRAT or titanium basic structure. This application will produce the least-weight thrust structure.

One of the inherent problems with a fuel or propellant transfer system is in the sequencing of the transfer/shutoff valves. To obviate this problem, a tank-to-tank direct transfer system has been devised, Figure 4.3-10.

Transfer of the LH_2 occurs at the forward structural interface with the transfer line being supported from the attach structure. Because of the small pressure head between the booster and orbiter mounted LH_2 tanks, there is a requirement for a boost/transfer pump. This pump is located between the LH_2 supply in the booster and the propellant shutoff valve below the disconnect.

A quick-disconnect, or coupling, is located at the interface plane and is designed for zero leakage on separation. A shutoff valve is installed between the orbiter mounted coupling half and the orbiter tank.

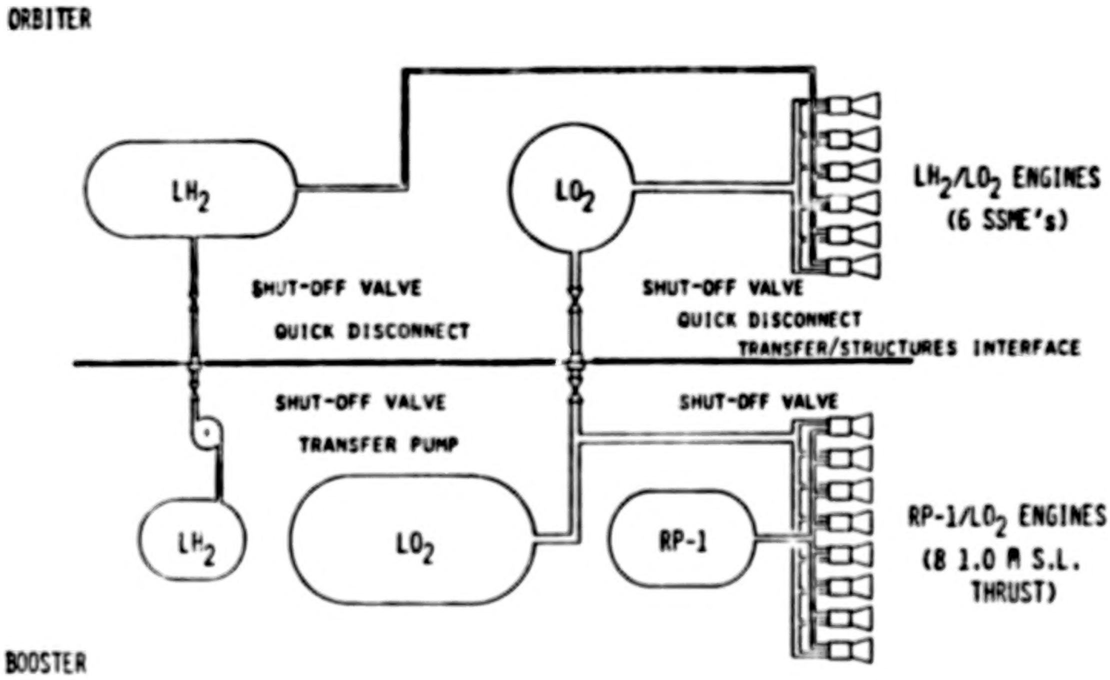


Figure 4.3-10. Ascent Propellant Transfer Schematic

Transfer of the LO_2 occurs at the aft interface and is accomplished by the normal tank pressure head on the booster tank. The booster tank is located substantially higher than the orbiter LO_2 tank, thus augmenting the pressure head with a gravity head. Mechanization of the LO_2 transfer is identical to the LH_2 system with the exception of not requiring a boost/transfer pump.

Both the LO_2 and LH_2 connections to the quick-disconnect have a six-degree-of-freedom capability to accommodate misalignments and differential deflections during ascent, as well as thermal contraction during fueling operations.

While major components of the fuel crossfeed system are derived from the Shuttle system design, the LH_2 transfer system adopted for the present HLLV system requires the addition of a boost pump and highly flexible joints to accommodate the substantial differential contraction and expansions which will be experienced during fueling operations. The forward LH_2 transfer point was incorporated to minimize the length of large-diameter cryogenic plumbing required for the launch system.

The large scale of the vehicles will require long structural elements for the thrust structure components. These lengths will likely be so long as to incur significant weight penalties to reduce column bending. Alternate approaches, such as the inclusion of intermediate thrust structure bulkheads, may be more efficient weight-wise.

The single-point thrust transfer assumed for this point design study may induce higher load concentrations in either or both vehicles than the structural systems may be able to efficiently absorb. This concern could be resolved through detailed design studies and modification of the present system or by the application of some alternate concept.

The very high fuel flow rates required by the ascent propulsion system will necessitate large-diameter cryogenic plumbing. Designs approaching these are intrinsic with the Shuttle orbiter and external tank, although smaller in diameter. However, the LH_2 pump identified for the present concept will require a major design study. This study should include trades to determine the relative benefits of overpressurizing the booster LH_2 tank to effect the necessary fuel transfer head.

An extensive ground test program of several candidate LH_2 transfer systems will be required to verify the performance efficiency of each. The results of these tests will identify the design approach which appears to provide the most favorable and reliable design.

4.3.4 PRELIMINARY THERMAL/STRUCTURAL REQUIREMENTS

This final technical issue addresses that area where perhaps the greatest amount of basic technology development is required. During the early recoverable launch vehicle studies, substantial effort was expended to develop materials, structural systems, and thermal protection systems capable of withstanding the severe entry heat loads typical of lifting entry and yet be fully reusable without major inspection or refurbishment. This activity has since all but ceased with the definition of the Space Shuttle TPS since no other requirements then existed. Some small-scale development has continued, however, but not at the level required for another new system development.

The thermostructural system is the key to the entry "survivability" of any entry system. The present design challenge is not only to meet this basic design objective but, at the same time, to create a thermostructural system which is capable of "airline operation"; that is, for the system to be essentially removed from the realm of flight-to-flight maintenance. Ideally, post-flight operations should be restricted to flight preparation, payload installation, and refueling, with thermostructural system inspection and maintenance relegated to scheduled IRAN cycles.

The achievement of these goals will require extensive analysis and development at all levels of design—basic materials through to total structural system design and verification.

Reentry design requirements are specified by vehicle wing loading with payload onboard (abort once around). Reentry trajectories were developed for HLLV utilizing basic Space Shuttle aerodynamics with a wing loading of 400 kg/m^2. Associated with these trajectories was a control system that closely approximated that of the Shuttle. Results of a candidate trajectory are shown in Figure 4.3-11 for altitude, nose temperature, heat rate, and angle of attack as functions of velocity. During an entry of this type, bank angle modulation controls maximum heat rate and deceleration. At lower velocities, both angle of attack and bank angle are modulated to null range errors and control touchdown point.

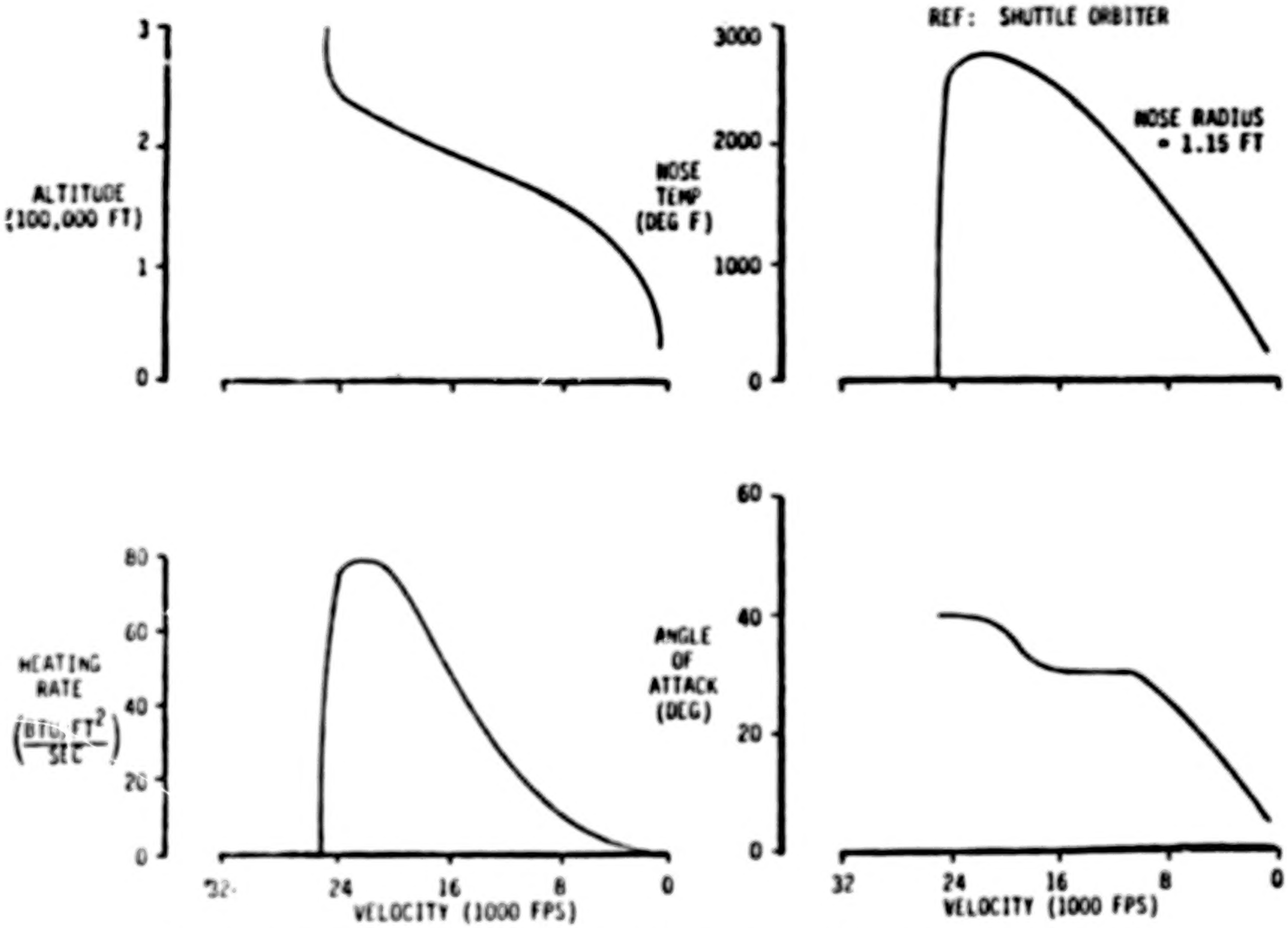


Figure 4.3-11. Orbiter Reentry Trajectory Time History

The SPS orbiter maximum radiation equilibrium isotherms are presented in Figure 4.3-12. The orbiter is pitched to high angle of attack (35° to 40°) during vehicle entry to minimize aerothermodynamic heating. Therefore, the upper surface temperature predictions are conservative since conventional attached flow heat transfer methodology tends to be conservative in such regions. Based upon these temperature predictions, a conceptual TPS design of the reusable insulation type of concept (Shuttle orbiter) with improved tile and/or direct bond will be adequate for the design mission. Advanced metallic thermostructures will require substantial development to meet post-flight minimum refurbishment (i.e., none) requirements.

The booster maximum radiation equilibrium isotherms (constant temperature lines) have been analytically determined. As shown in Figure 4.3-13, the peak aerodynamic heating occurs during booster flyback while the vehicle is subjected to an angle of attack of approximately 25°. Most of the vehicle upper surface lies in the separated flow region in which conventional attached flow heat transfer methodology tends to be conservative. The results presented do not consider the effects of (1) booster exhaust plume recirculation induced heating nor (2) heat transfer amplification due to clustered bodies interaction. These impacts can only be evaluated through a coordinated ground test program. Thus (with these considerations in mind), the SPS booster can conceivably be built with late 1980's technology using hot structures.

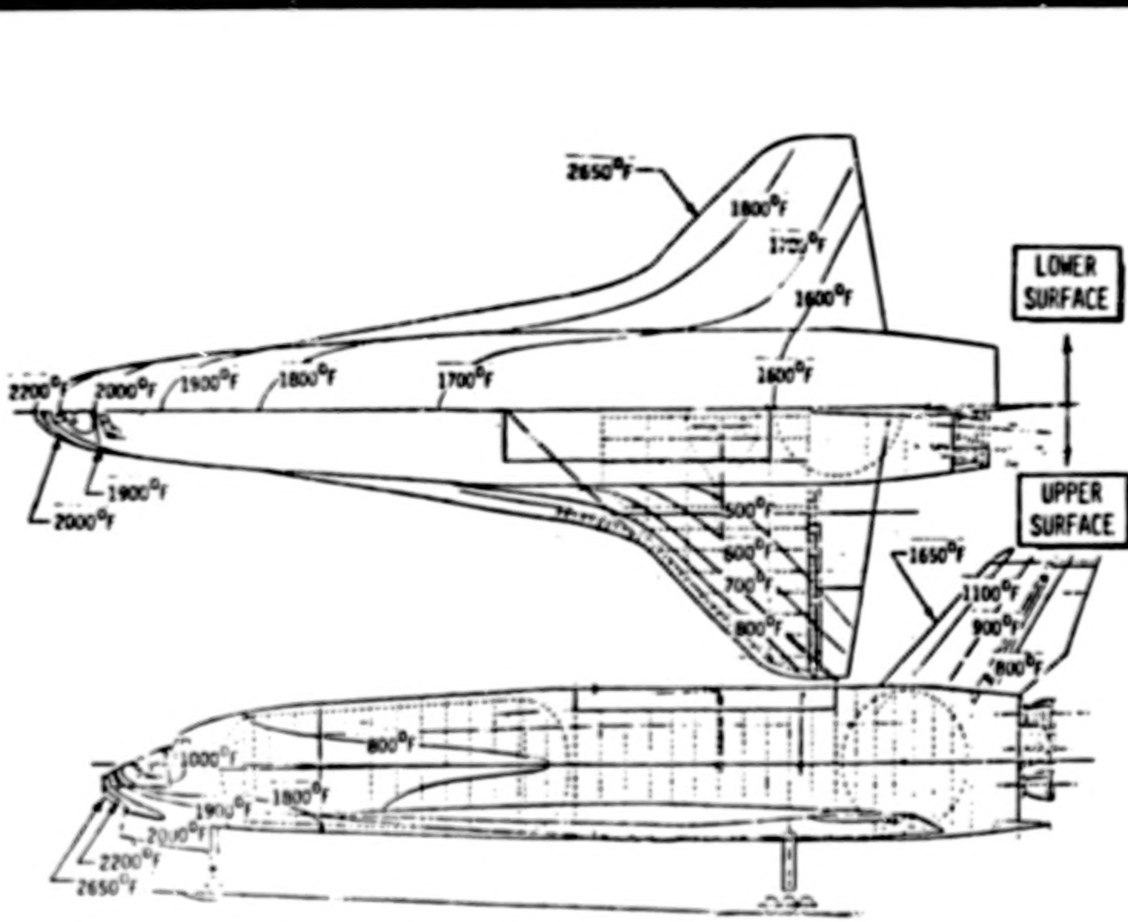


Figure 4.3-12. SPS Orbiter Maximum Radiation Equilibrium Isotherms

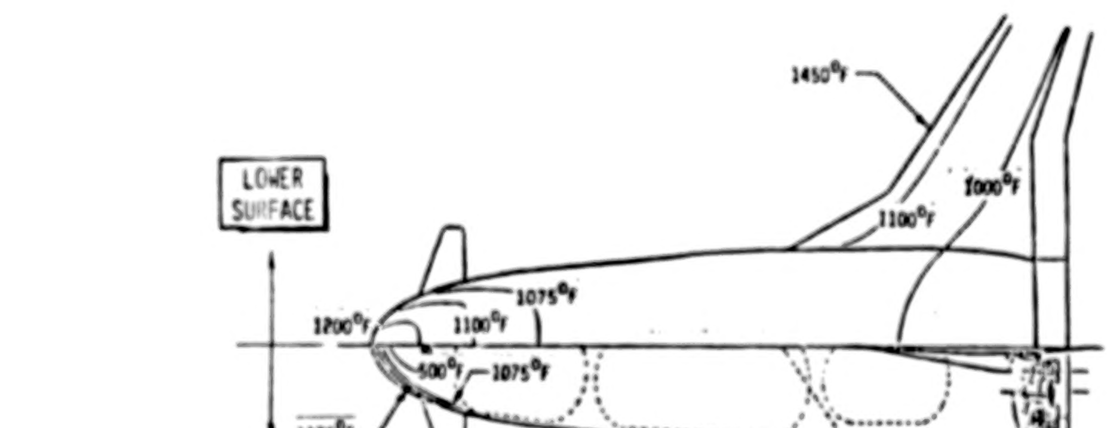


Figure 4.3-13. SPS Booster Maximum Radiation Equilibrium Isotherms

Point design definitions for the thermostructural system of the present launch vehicles were made to enable the preliminary development of the vehicle concepts for evaluation. These selections were based on the expected availability and capability of known material and structural concepts. Neither selection, for the orbiter nor the booster, is considered anywhere near firm. Very substantial development is required before any such decision can be made with any validity; design verification is a stringent requirement.

The application of graphite-polyimide primary structure was selected since there is currently considerable emphasis being placed on the development of this material by both government and industry. Trade studies have been conducted on the Shuttle orbiter which indicate that substantial weight savings can be realized by reduced TPS thickness requirements, since the GR/PI material can accept TPS backface temperatures up to 600°F now and perhaps up to 800°F with further development. NASA/Ames-developed FRCI tiles form most of the external TPS on the present orbiter. These may be direct-bonded to the GR/PI tiles and are reported to be far more resistant to foreign object damage.

Non-integral tanks have been selected for both the booster and orbiter cryogenic fuel systems. Integral tanks may be preferable, but have been rejected in the past because maintenance and inspection with the requisite internal insulation has not been possible on a routine basis.

The launch system is designed to stage at a velocity which will permit the application of a heat sink all-metallic booster structure. Extensive use of metal matrix materials is expected to be representative of early 1990's technology and has been assumed for the present booster design.

Representative sections of structure from both the booster and the orbiter are illustrated in Figure 4.3-14. The orbiter structure utilizes graphite-polyimide materials as the primary structure. NASA/Ames-developed FRCI tiles are bonded directly to the substrate since the coefficients of expansion of each material are nearly identical, thus requiring no strain isolation layer as in the Shuttle orbiter TPS design. Alternate approaches which may be made feasible with substantial advanced development include all-metallic structure and the application of localized active cooling. The former will require considerable development to identify materials capable of repeatably withstanding high-temperature environments without degradation of their properties and not requiring fragile surface coatings. The latter alternate has been shown to be effective, but does require a complex fluid system.

The booster assumes the application of advanced metal matrix hot structure. Fiber reinforced advanced titanium (FRAT) is shown on the conceptual section provided by the Rockwell North American Aircraft Division in support of the present study. Other materials than titanium may ultimately be applicable but, again, will require considerable preliminary development.

The final structural system to be ultimately selected must satisfy requirements which are basically the same as the structural systems in use today. The significant change, of course, is in the operating environment. This environment is much more stringent than is experienced today in any aircraft-like systems other than the Shuttle orbiter. It will eventually require the

establishment of new or highly modified specifications to augment those in use today.

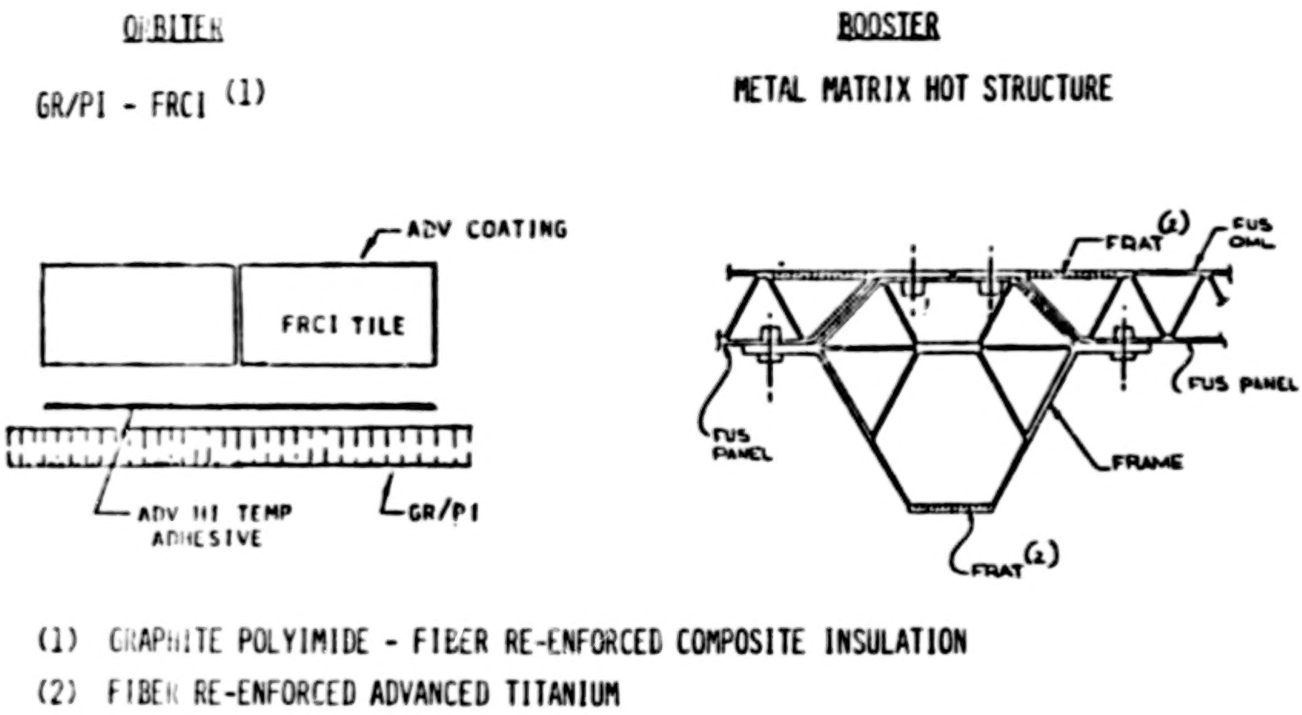


Figure 4.3-14. Typical Skin Concepts

The materials which have been identified to date as being capable of withstanding the high temperatures of entry are generally very fragile, expensive, heavy, or rare, in any combination. Those materials which will have application to future lifting entry vehicles will have to overcome these shortcomings to be acceptable for the routine operations contemplated for the SPS HLLV transportation system.

Table 4.3-1 was extracted from an AFFDL technical report published over ten years ago (AFFDL-TR-69-94). Comparison of the materials and their properties cited in this table with a similar list of today's materials reveals little change—indicative of relatively low developmental activity during the intervening years.

Multi-wall TPS concepts have been suggested for application to orbiter structural systems. It consists essentially of thin-wall material expanded to a low-density metallic structure exhibiting low conductivity, Figure 4.3-15. The concept is made possible by the development of superplasticforming/diffusion bonding (SPF/DB) techniques pioneered by Rockwell. The truss core sandwich may be integrally formed with the multi-wall TPS in a single manufacturing operation. The temperature requirement for the inner and outer faces defines the multi-wall thickness requirements.

In order to meet the postulated operational requirements of the SPS transportation system, the cryogenic tanks of both the booster and the orbiter must

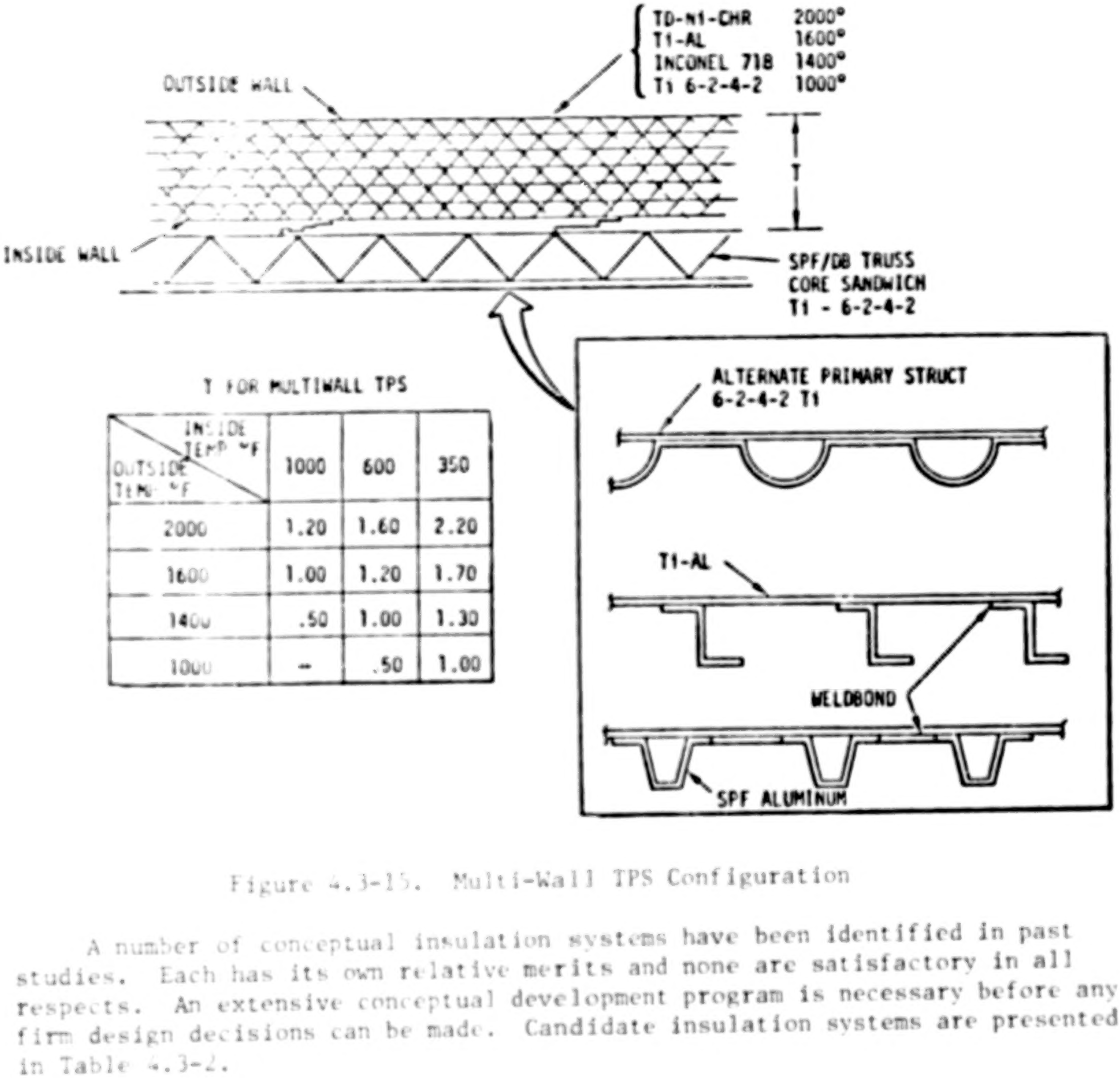
Table 4.3-1. Merit Indexes for Candidate Materials

REF: AFFDL-TR-69-94

TEMP RANGE	VEHICLE STRUCTURAL APPLICATION	CANDIDATE MATERIAL (1)	PHYSICAL PROPERTIES (3) P S G E	TENSILE PROPERTIES (3) P S G E	CREEP RESISTANCE (2)(3) P S G E	FORMABILITY (3) P S G E	WELDABILITY (3) P S G E	OXIDATION RESISTANCE (3) P S G E	LEADING CANDIDATE MATERIALS	REMARKS
475° - 1300°F	WING TANKS BODY STRUCTURE	2219 T87	G	G	G	G	G	E	*2219 T87	GOOD FABRICABILITY AND STRENGTH
		6061 T6	G	S	S	E	E	E		GOOD WELDABILITY, MODERATE STRENGTH
		7075 T6	G	E	G	G	P	E	*7075 T6	HIGH STRENGTH, WELDING NOT PRACTICAL
		BE 38A1	E	S	S	P	NA	P		NOT WELDABLE
		AMS 7902	E	S	S	P	NA	P		
		8A1 1MO 1V	G	G	G	S	G	E	8A1 1MO 1V	GOOD STRENGTH AND FABRICABILITY
		6A1 4V	G	G	G	S	G	E		GOOD STRENGTH AND FABRICABILITY
		5A1 2.5Sn	G	S	S	G	G	E		LOW STRENGTH, GOOD FABRICABILITY
1300° - 2000°F	UPPER SURFACE PRIMARY AND SECONDARY STRUCTURE HEAT SHIELD	COBALT BASE ALLOY								
		HAYNES 25	P	S	G	E	S	G	*HAYNES 25	ANNEALED MATERIAL WITH MODERATE TENSILE PROPERTIES. GOOD OXIDATION RESISTANCE TO 1600°F
		NICKEL BASE ALLOY								
		INCO 625 (1400°F)	P	S	G	G	S	G	*INCO 625	MATRIX STRENGTHENED ALLOY WITH MODERATE TENSILE PROPERTIES. METALLURGICALLY UNSTABLE ABOVE 1400°F
		INCO 718 (1400°F)	P	E	G	G	G	G		AGE HARDENABLE ALLOY WITH HIGH TENSILE PROPERTIES. MODERATE CREEP RESISTANCE
		HASTALLOY X	P	P	G	G	G	E		SUPERIOR OXIDATION RESISTANCE TO 2000°F
		RENE 41 (ANN) (1600°F)	P	E	E	S	S	G	*RENE 41	ANG WELDABILITY, SUBJECT TO EMBRITTLEMENT AND ALLOY DEPLETION ABOVE 1600°F
		TD NICKEL	P	P	P	S	P	S		NOT COMPETITIVE WITH MECHANICAL PROPERTIES OF OTHER SUPERALLOYS
		TD NiCr	P	S	S	S	P	E	*TD NiCr	CANDIDATE UNCOATED MATERIAL FOR APPLICATION TO 2200°F
2000° - 2500°F	LOWER SURFACE LEADING EDGE AND HEAT SHIELD	CHROME 30	P	P	S	P	NOT PRACTICAL	E		EXTREMELY BRITTLE MATERIAL AT ROOM TEMPERATURE
		TD NICKEL	P	P	S	S	P	S		NOT TO BE USED AS PRIMARY STRUCTURAL APPLICATION, NOT COMPETITIVE WITH Cb ALLOYS
		TD NiC (2200°F)	P	S	S	S	P	E	*TD NiC	CANDIDATE UNCOATED MATERIAL FOR APPLICATION TO 2200°F
		COLUMBIUM (4) ALLOY								
		D 43	S	G	E	G	S	S		POOR WELDABILITY
		B 66	S	E	G	S	P	S		SUPERIOR DENSITY COMPENSATED STRENGTH VALUES, POOR FORMABILITY AND WELDABILITY PROPERTIES
		FS 85	S	G	E	G	G	S		HIGH DENSITY
		C 129Y	S	G	S	G	S	S		EXCEPT FOR LOWER CREEP RESISTANCE, SIMILAR TO Cb 752
		Cb 752	S	G	G	G	S	S	*Cb 752	MODERATELY HIGH MECHANICAL PROPERTIES
2500° - 3500°F	LEADING EDGE	TANTALUM (4) ALLOY								
		90Ta 10W	S	G	G	E	G	S	*90Ta 10W	MODERATE MECHANICAL PROPERTIES, VERY GOOD WITH RESPECT TO FABRICABILITY
		T 222	S	E	G	G	S	S		HIGH MECHANICAL PROPERTIES, MILL ROLLING PROBLEMS
		T 111	G	E	E	P	NA	S		
3500° - 4000°F	NOSE CAP	TUNGSTEN THORIA	G	P	NA	P	NA	E	*TUNGSTEN THORIA	TESTED IN REENTRY PROFILE
		TUNGSTEN	G	P		S	(4)	S		LIMITED BY OXIDATION PROTECTIVE SYSTEM
		ZIRCONIA ROD	S	P		P		E		TESTED IN REENTRY PROFILE

NOTES (1) MAXIMUM STRUCTURAL TEMPERATURE LIMIT
(2) BASED ON 0.5 PERCENT CREEP AT SPECIFIED TEMPERATURE
(3) RATING LEGEND IS AS FOLLOWS
E - EXCELLENT
G - GOOD
S - SATISFACTORY
P - POOR
(4) WITH OXIDATION PROTECTIVE COATING
NA - NOT APPLICABLE

be designed such that they require little or no inspection outside normal maintenance cycles. Similar requirements are placed on the tank insulation also. The tanks of the present point design are identified as being non-integral structurally with the tank insulation system, permitting relatively easy inspection when required but not allowing the buildup of icing on the external surfaces nor any cryopumping.



T FOR MULTIWALL TPS

OUTSIDE TEMP °F \ INSIDE TEMP °F	1000	600	350
2000	1.20	1.60	2.20
1600	1.00	1.20	1.70
1400	.50	1.00	1.30
1000	–	.50	1.00

Figure 4.3-15. Multi-Wall TPS Configuration

A number of conceptual insulation systems have been identified in past studies. Each has its own relative merits and none are satisfactory in all respects. An extensive conceptual development program is necessary before any firm design decisions can be made. Candidate insulation systems are presented in Table 4.3-2.

As indicated earlier, substantial work is required before any firm decisions regarding thermostructural concepts for either the booster or the orbiter can be made. Potential problems exist in nearly all related areas at present. These include the development of metallic thermostructural materials, where significant basic technology effort is required, insulation concepts for the cryogenic tanks, and high thermal gradient structural systems.

Table 4.3-2. Cryogenic Insulation Material Systems

<u>Evacuated System</u>

- Compressed superinsulation

<u>Purged Systems</u>

- Quartz fiber purged with helium
- Marshield purged with helium
- Aluminum shields with dimpled fiberglass spacers
- Polyurethane foam

<u>Sealed Systems</u>

- Polyurethane foam (Freon or CO_2 blown)
- Mylar honeycomb sandwich filled with polyurethane foam
- Phenolic fiberglass honeycomb sandwich
- Corkboard
- Polyimide foam

<u>Sealed and Purged Systems</u>

- The sealed systems are used with a quartz-fiber blanket purged with helium or nitrogen

The materials specified for the outer layers of the orbiter TPS must withstand an extreme thermal and stress environment. Those materials available today which can meet some of these requirements do not meet all of the desired criteria—coatings are subject to foreign object damage, embrittlement occurs after repeated exposure to high temperature environments reducing the physical strength of the material, and the materials are heavy, costly, or in very short supply, etc.

Cryogenic tank insulation requires extensive development before design selections can be made with confidence.

Coupling hot structures to relatively cold substructures, such as crew cabin walls or cryogenic tanks, requires additional technology development before the integrity of such joints can be assured.

Substantial materials and basic metallurgical development must be undertaken to identify or formulate materials which can meet the requirements of the entry thermal environment and yet be readily available, serviceable, etc. Active cooling systems are feasibly, but their "reasonability" is subject to question; the distribution system which is normally integral with the vehicle skin is a direct beneficiary of the SPF/DF process since the former need for extensive tube welding has been eliminated.

The development of thermostructural systems capable of taking full advantage of the potentially available advanced materials must also be intensively pursued. A wide variety of candidates is already available, but the relative merits of each need to be determined.

All aspects of cryogenic tank design must be evaluated and resolved. This includes the analysis of integral and non-integral tanks, insulation techniques, and operational utility.

Finally, basic techniques for the prediction of heating rates must be refined and verified.

All of the analyses and developments cited above must be verified through extensive cyclic testing under simulated operational conditions. Additional testing to qualify production techniques (as opposed to laboratory techniques) must be conducted to establish the producibility of the more promising materials and fabrication techniques.

5.0 ELECTRIC ORBITAL TRANSFER VEHICLE

The rationale for electric OTV selection over the conventional chemical systems is clearly illustrated in Figure 5.0-1. Because of the limited specific impulse of chemical rocket systems (i.e., <500 sec), the mass to low earth orbit requirement is increased approximately three-fold due to chemical propellant requirements. Also indicated, is a comparison of mass to orbit requirements for a chemical attitude control system (CACS) versus an electric thruster attitude control system (EACS). Again, a decreased mass to orbit (i.e., ~25%) requirement is indicated for an EACS. Since transportation costs from earth to LEO is the prime contributor to overall SPS transportation cost, the electric system offers a considerable cost advantage over chemical systems.

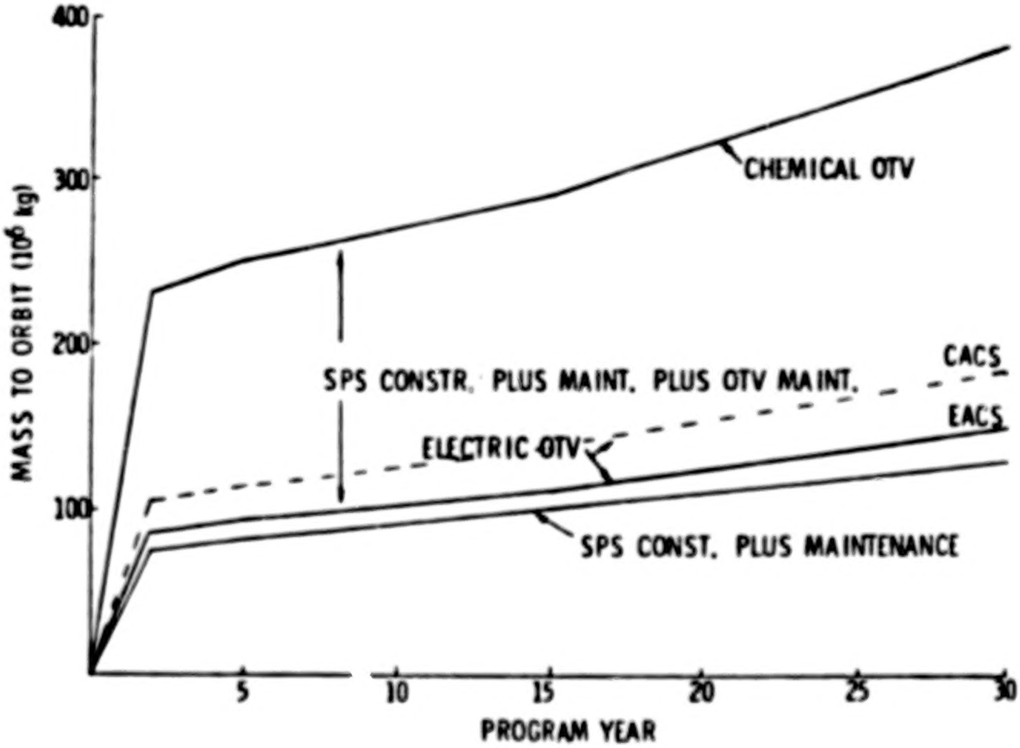


Figure 5.0-1. Mass-To-Orbit Requirements

The basic EOTV concept was developed during the Exhibit C studies, and that configuration is included for reference. An EOTV configuration update was deemed necessary because of changes in the reference satellite concept and a need to reduce the maximum allowable thruster beam current density to assure adequate thruster grid life. The configuration update led to a similar configuration with 20% fewer (but larger diameter) thrusters, and a 30% increase in payload weight with essentially the same orbital burden factor.

5.1 EXHIBIT C REFERENCE EOTV CONCEPT

The Exhibit C reference EOTV concept is illustrated in Figure 5.1-1, and a weight performance summary is given in Table 5.1-1. The thruster operating characteristics are presented in Table 5.1-2.

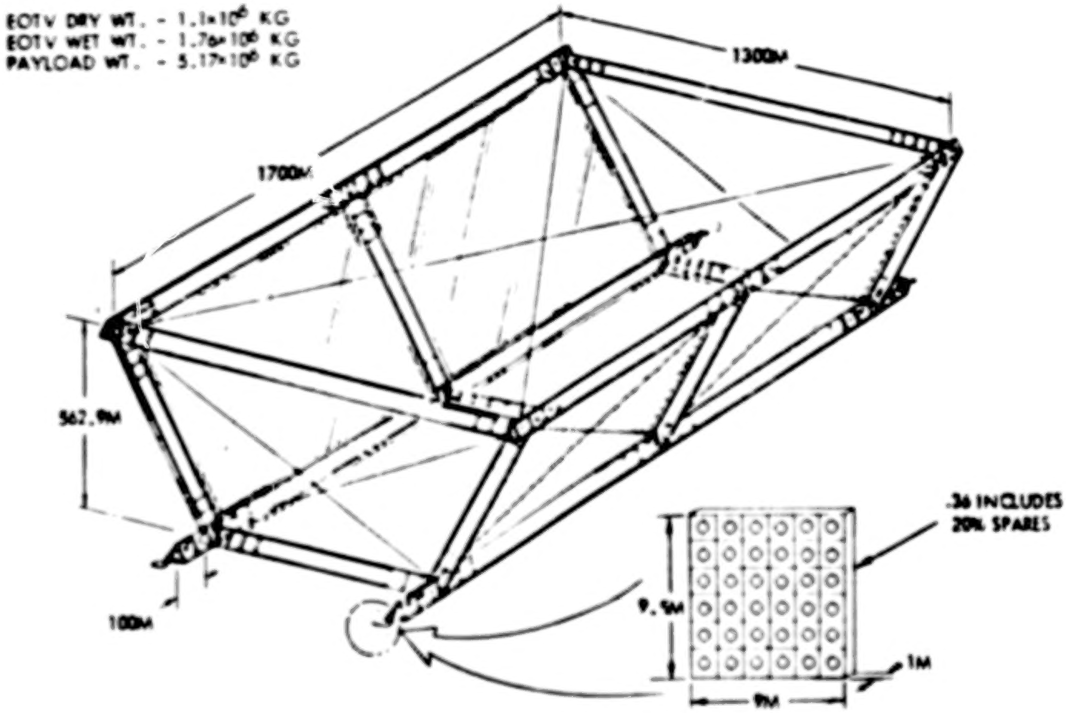


Figure 5.1-1. Selected EOTV Configuration

Table 5.1-1. EOTV Weight/Performance Summary (kg)

SOLAR ARRAY		588,196
CELLS/STRUCTURE	299,756	
POWER CONDITIONING	288,440	
THRUSTER ARRAY (4)		96,685
THRUSTERS/STRUCTURE	10,979	
CONDUCTORS	4,607	
BEAMS/GIMBALS	2,256	
PROPELLANT TANKS	78,843	
ATTITUDE CONTROL SYSTEM		186,872
POWER SUPPLY	184,882	
SYSTEM COMPONENTS	274	
PROPELLANT TANKS	1,716	
EOTV INERT WEIGHT		871,753
25% GROWTH		217,938
TOTAL INERT WEIGHT		1,089,691
PROPELLANT WEIGHT		666,660
TRANSFER PROPELLANT	655,219	
ACS PROPELLANT	11,441	
EOTV LOADED WEIGHT		1,756,351
PAYLOAD WEIGHT		5,171,318
LEO DEPARTURE WEIGHT		6,927,669
PROPELLANT COST DELIVERED ($/KG P/L)		4.72

Table 5.1-2. EOTV Thruster Characteristics

- MAXIMUM OPERATING TEMPERATURE - 1900° K
- TOTAL VOLTAGE - 8300 VOLTS
- GRID VOLTAGE - 2000 VOLTS MAXIMUM
- BEAM CURRENT - 1887 AMP
- SPECIFIC IMPULSE - 8213 SEC
- THRUSTER DIAMETER - 76 CM
- THRUST/THRUSTER - 69.7 NEWTON
- NUMBER OF THRUSTERS - 144 (INCLUDES 25% SPARES)
- MAXIMUM OF 64 THRUSTERS OPERABLE SIMULTANEOUSLY

5.2 EOTV CONFIGURATION UPDATE

The electric orbital transfer vehicle concept, Figure 5.2-1, is based on the same construction principles of the GaAs reference satellite configuration. The commonality of the structural configuration and construction processes with the satellite design is evident. The structural bay width of 700 m (solar array width of 650 m) is the same as that of the satellite. The structural bay length is reduced from 800 m to 750 m for compatibility with the lower voltage requirement of the EOTV. The concept utilizes electric argon ion thruster arrays.

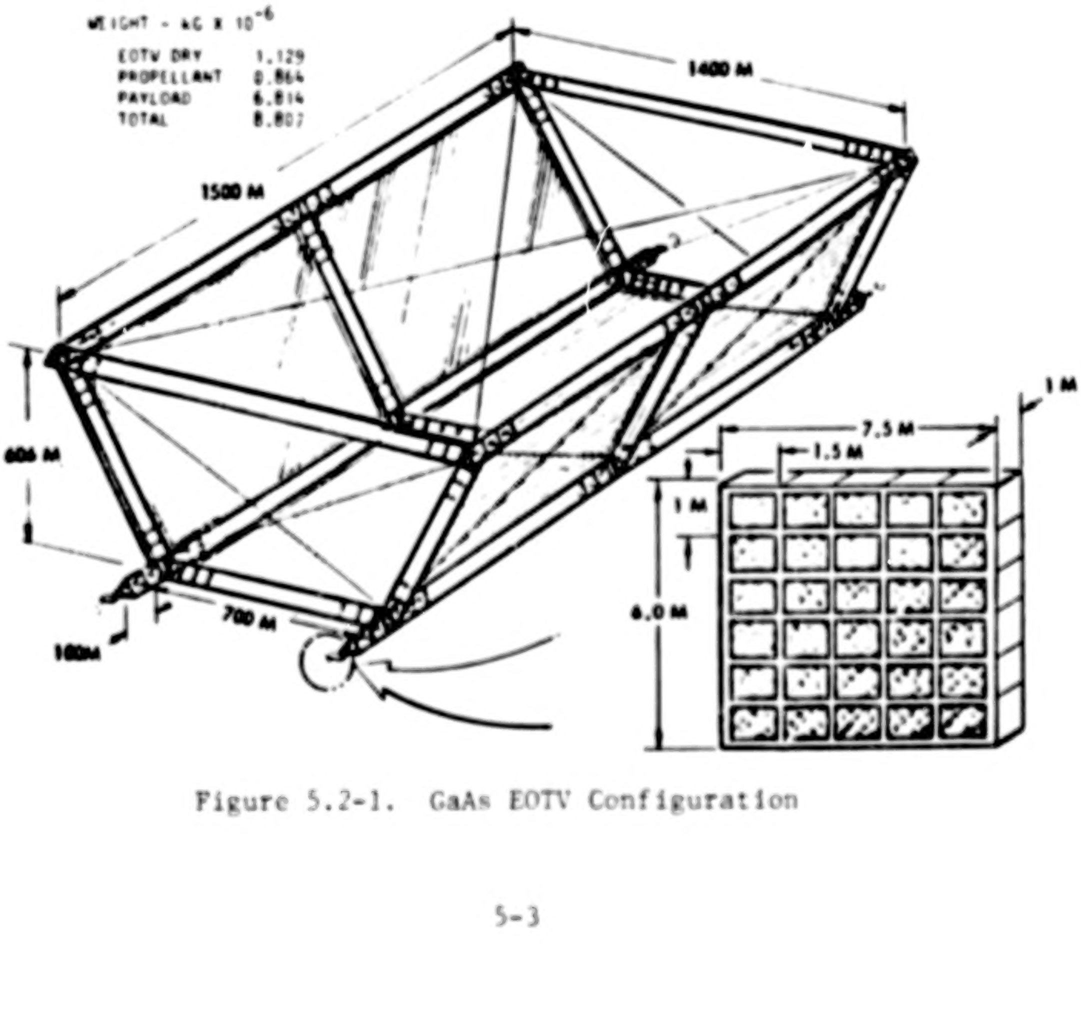


Figure 5.2-1. GaAs EOTV Configuration

The primary assumptions used in EOTV sizing are essentially the same as those employed during the Exhibit C study phase and are summarized in Table 5.2-1. The orbital parameters are consistent with SPS requirements and the delta-V requirement was taken from previous SEP and EOTV trajectory calculations. A 0.75 delta-V margin is included in the figure given.

Table 5.2-1. EOTV Sizing Assumptions

- LEO ALTITUDE - 487 KM @ 31.6° INCLINATION
- SOLAR INERTIAL ORIENTATION
- LAUNCH ANY TIME OF YEAR
- 5700 M/SEC ΔV REQUIREMENT
- SOLAR INERTIAL ATTITUDE HOLD ONLY DURING OCCULTATION PERIODS
- 50° PLUME CLEARANCE
- NUMBER OF THRUSTERS - MINIMIZE
- 20% SPARE THRUSTERS - FAILURES/THRUST DIFFERENTIAL
- PERFORMANCE LOSSES DURING THRUSTING - 5%
- ACS POWER REQUIREMENT - MAXIMUM OCCULTATION PERIOD
- ACS PROPELLANT REQUIREMENTS - 100% DUTY CYCLE
- 25% WEIGHT GROWTH ALLOWANCE

During occultation period, attitude hold only is required (i.e., thrusting for orbital change is not required).

Since thruster grid changes are assumed after each mission, a minimum number of thrusters are desired to minimize operational requirements.

An excess of thrusters are included in each array to provide for potential failures and primarily to permit higher thrust from active arrays when thrusting is limited or precluded from a specific array due to potential thruster exhaust impingement on the solar array or to provide thrust differential as required for thrust vector/attitude control. A 5% specific impulse penalty was also applied to compensate for thrust cosine losses due to thrust vector/attitude control.

An all-electric thruster system was selected for attitude control during occultation periods. The power storage system was sized to accommodate maximum gravity-gradient torques and occultation periods. A very conservative duty cycle of 100% was assumed for establishing ACS propellant requirements. A 25% weight growth margin was applied as in the case of the SPS.

The solar array size is dictated primarily by the requirement to maintain the same construction approach as the satellite, consistent with specific EOTV voltage requirements. The solar array voltage must be as high as possible to reduce wiring weight penalties and to provide high thruster performance; yet, power loss by current leakage through the surrounding plasma must be minimized. At the proposed LEO staging base, with very large solar arrays and high efficiency cells, an upper voltage limit of 2000 volts is postulated.

Since GaAs solar cells are employed in this concept with a concentration ratio of 2 on the solar cell blanket, the resulting cell operating temperature of 125°C allows continuous self-annealing of the solar cells during transit through the Van Allen radiation belt.

The solar blanket width of the satellite (650 m) is retained for the EOTV. A blanket length (per bay) of 1400 m is determined by the solar cell string length required to achieve the desired operational conditions of 2000 V (string length of approximately 63.5 m). Eleven such strings result in a solar blanket length of approximately 700 m. Twenty-five meters of additional structural length at each end of the solar blanket are required to provide for catenary support. These considerations led to the selection of a two-bay configuration with structural dimensions of 700×1500 m (solar blanket size, 650×1400 m) with a total power output of 309 MW (includes 6% line losses).

The solar array weights were scaled from satellite weights, and are summarized in Table 5.2-2.

Table 5.2-2. EOTV Solar Array Weight Summary (10^{-6} kg)

Structure		0.095
Primary	0.041	
Secondary	0.054	
Mechanisms		0.004
Concentrators		0.033
Solar panels		0.229
Power distribution and controls		0.262
Maintenance provisions		0.003
Information management		0.002
	Total	0.628

Having established the solar array operating voltage, the maximum screen grid voltage is established which, in turn, fixes propellant ion specific impulse. In order to assure adequate grid life, to assure a minimum roundtrip capability of approximately 4000 hours, a maximum beam current of 1000 A/m^2 was selected. Based on the available power and a desire to maintain reasonable thruster size, the remaining thruster parameters are established. A rectangular thruster configuration (1×1.5 m) is assumed. Primary thruster characteristics are summarized in Table 2.2-4.

Based on the individual thruster power requirements and the available array power, 100 thrusters may be operated simultaneously. An additional 20 thrusters are added to provide a thrust margin when thruster array orientation might preclude firing due to potential ion impingement on the solar array. The thrusters are arranged in four arrays of 30 thrusters each. The thruster array mass summary is presented in Table 5.2-4.

Table 5.2-3. Argon Ion Thruster Characteristics

Maximum total voltage, volt	4405
Maximum operating temperature, °K	1330
Screen grid voltage, volt	1880
Accelerator grid voltage, volt	-2525
Beam current, amp	1500
Beam power, watt	2.82×10^6
Specific impulse, second	7963
Thrust, newton	56.26

Table 5.2-4. Thruster Array Mass Summary (kg)

Thrusters and structure	24,000
Conductors	6,000
Beams and gimbals	2,200
Power processing	2,000
Attitude reference system	1,000
Batteries and charger	154,000
Total	189,200

The EOTV performance is based on a 120-day trip time from LEO to GEO (obtained from trade studies). Knowing the propellant consumption rate of the thrusters and the thrusting time, the maximum propellant which can be consumed is determined which, in turn, defines the payload capability. The vehicle is also sized to provide for the return to LEO of 10% of the LEO-to-GEO payload. The EOTV weight summary is presented in Table 5.2-5.

Table 5.2-5. EOTV Mass Summary (10^{-6} kg)

Solar array		0.628
Thruster array (4)		0.189
Propellant tanks and dist.		0.086
EOTV (dry)		0.903
Growth (25%)		0.226
EOTV, total		1.129
Propellant		0.864
Main LEO-GEO	0.655	
Main GEO-LEO	0.143	
Attitude control	0.065	
EOTV (wet), total		1.993
Payload		6.814
LEO departure		8.807
GEO arrival		8.116
GEO departure		1.971
LEO arrival		1.822

6.0 TRANSPORTATION SYSTEM OPERATIONS/TECHNOLOGY REQUIREMENTS

A primary objective was to identify key elements of transportation system design requirements/characteristics and attendant operations requirements which would enhance overall SPS system operations and costs. Initial efforts were directed toward the identification of major transportation system cost elements and the definition of design and operational features that could reduce those costs along with the technology advancement requirements needed to implement those design and operational features. Although all SPS transportation elements were addressed, primary emphasis was placed on earth-to-LEO transportation, the HLLV.

The Exhibit C studies also showed a significant cost impact by using the PLV throughout the SPS program for transfer of personnel from earth to LEO. An analysis of using the alternative approach of imposing the requirement for transfer of personnel by the SPS-HLLV was also evaluated and the results are reported in the next section of this report and volume Volume VI, Cost and Programmatics.

6.1 GROUND OPERATIONS DEFINITION

The major element of ground operations are related to launch vehicle turnaround requirements. The high launch frequency demands an airline operations concept which in turn dictates vehicle design requirements which will result in the near-elimination of post-flight refurbishment and checkout other than that required for payload installation, mating and fueling. A summary of primary turnaround operations are presented in Table 6.1-1 and some of the key vehicle design requirements are summarized in Table 6.1-2. It is noted that turnaround time, in itself, would effect required vehicle operating fleet size which would have a minimal cost impact. However, the prime objective of reducing turnaround time is to maintain a "hands-off" policy which will minimize servicing crew requirements.

The key operational technology requirements of the HLLV are in the areas of:

- Structural/Thermal Protection Systems
- Propellant Tank Insulation Systems
- Liquid Rocket Engine/Component Life
- Self Monitoring/Diagnostic Systems

The materials required for the exterior of the vehicle must repeatably withstand an extreme thermal and stress environment. The materials available today which are capable of meeting some of these requirements cannot meet all of the desired criteria: coatings are subject to foreign object damage; embrittlement occurs after repeated exposure to environments resulting in reduced physical strength; the materials are heavy, costly, and/or in short supply. The development of thermostructural systems capable of taking full advantage of the potentially available advanced materials must be pursued. A wide variety of candidates

Table 6.1-1. Summary of Ground Turnaround Operations

MAJOR COST/TIME DRIVERS	IMPACT ON TURNAROUND OPERATIONS
1. SYSTEM SAFING & DESERVICING	SAFING, DESERVICING, & PURGING OF MAIN ENGINES AND FUEL MANAGEMENT SYSTEM
2. INSPECTION & DAMAGE IDENTIFICATION	IDENTIFY DAMAGE, MALFUNCTIONS, AND SUPPORT FAULT ISOLATION
3a. SCHEDULED MAINTENANCE (REFURBISHMENT	REMOVE/REPLACE EXPENDED HARDWARE, LIMIT-LIFE LRU, TIME-CYCLE HARDWARE & ENGINE TEST
3b. UNSCHEDULED MAINTENANCE (CORRECTIVE)	REMOVE/REPLACE MALFUNCTIONS & ANOMALIES NOTED FROM ON-BOARD CHECKOUT & FAULT ISOLATION SYS.
4. THERMAL PROTECTION SYSTEM REFURBISHMENT	REMOVE/REPLACE PORTIONS OF DAMAGED TPS METALLIC THERMAL PANELS
5. LAUNCH PAD REFURBISHMENT	INERTING, SECURING, FUEL FLUSH/PURGE, DAMAGE REPAIR, & STRUCTURE VALIDATION
6. PAYLOAD BAY DOORS OPENING/CLOSING	GSE SUPPORT OF OPENING/CLOSING 0-g DOORS IN A 1-g ENVIRONMENT WITHOUT SEAL DAMAGE
7. PERSONNEL/CARGO MODULE INSTALLATION	INSTALLATION/REMOVAL & CHECKOUT OF PERSONNEL & CARGO MODULE
8. STAGE PROCESSING FACILITIES	STAGE PROCESSING, SUBSYSTEM VERIFICATION/CHECKOUT, AND PAYLOAD MODULE INSTALLATION
9. GSE & SUPPORT SYSTEMS OPERATIONS	SYSTEMS ACTIVATION/DEACTIVATION, GROUND CHECKOUT, AND FAULT DETECTION DATA PROCESSING
10. GROUND CHECKOUT AND SYSTEMS TESTS	CHECK OUT, VERIFY INTERFACES, REVERIFY FLIGHT SYSTEMS AND FUNCTIONAL TESTS
11. STAGES TRANSFERRED TO LAUNCH PAD	TOW TO PAD, RETRACT GEAR, ROTATE TO VERTICAL, HARD DOWN ON LAUNCH PAD
12. STAGES ATTACHED TO LAUNCH PAD	1ST & 2ND STAGES MATED TO PAD, INTERFACES CONNECTED, AND PRELAUNCH VERIFICATION TEST
13. 1ST & 2ND STAGES MATING AND INTEGRATION	MATE 1ST STAGE TO 2ND STAGE, INTERFACES CONNECTED, AND VEHICLE INTEGRATION TEST
14. LAUNCH READINESS VERIFICATION TEST	VERIFY END-TO-END FUNCTIONS, COMMUNICATIONS, AND READINESS STATUS CHECK
15. SERVICING/ON-PAD PROPELLANT LOADING	LOAD HAZARDOUS LIQUIDS, MAIN PROPELLANTS, AND CREW CONSUMABLES
16. FINAL LAUNCH OPERATIONS AND COUNTDOWN	STATUS VERIFICATION, ALL SYSTEMS FUNCTIONING, CREW INGRESS AND AVAILABLE LAUNCH WINDOW

Table 6.1-2. Summary of Transportation System Design Requirements

IMPACT AREA/DESIGN REQUIREMENT	DESIGN AND TECHNICAL IMPROVEMENTS
1. SYSTEM SAFING & DESERVICING	• DESIGN FOR AUTOMATIC VENTING, DUMPING & PURGING DURING ORBITAL REENTRY OPERATIONS • MINIMIZE SERVICE & GROUND CONNECTIONS FOR DESERVICING & PURGING
2. INSPECTION & DAMAGE IDENTIFICATION	• LOW-MAINTENANCE FEATURES DESIGNED INTO VEHICLE, GSE & FACILITIES • VEHICLE DESIGN INCLUDES USE OF ON-BOARD INTEGRATED DATA SYSTEM & MAINT. ACCESSIBILITY
3a. SCHEDULED MAINTENANCE (REFURBISHMENT)	• DESIGN FOR HIGH FLIGHT USAGE RATES TO REDUCE PARTS REPLACEMENT & MAINTENANCE • DESIGN IN-FLIGHT REDUNDANCY MGMT SYST TO USE REDUNDANT FUNCTIONAL PATHS AND MODES
3b. UNSCHEDULED MAINTENANCE (CORRECTIVE)	• MAX DESIGN USE OF ON-BOARD C/O, SYST MONITORING, FAILURE DETECTION, FAULT ISOLATION • PROVIDE FOR FAILURE TOLERANT DESIGN
4. THERMAL PROTECTION SYSTEM REFURBISHMENT	• DESIGN METALLIC TPS PANELS TO PROVIDE FOR QUICK-TURNAROUND-TIME REPLACEMENT
5. LAUNCH PAD REFURBISHMENT	• PROVIDE DESIGN FOR PROTECTIVE SHIELDING & DAMAGE REDUCTION TO FUEL CONN. & STRUCTURE • PROVIDE DESIGN TO MINIMIZE DAMAGE TO ACCESS ARMS, SERVICE TOWER, AND CREW EGRESS
6. PAYLOAD BAY DOORS OPENING/CLOSING	• DESIGN FOR OPENING/CLOSING OF 0-g DOORS IN 1-g ENVIRONMENT • DESIGN FOR BUILT-IN HYDRAULIC/PNEUMATIC OPENING SYST WITH GSE REDUNDANT SUPPORT
7. PERSONNEL/CARGO MODULE INSTALLATION	• SIMPLIFY DESIGN INSTALLATION REQUIREMENTS BY USE OF CONTAINERIZED CARGO CONCEPT • DESIGN INTERFACES LIMITED TO COMMUNICATIONS, INSTRUMENTATION & SECURING SYSTEM
8. STAGE PROCESSING FACILITIES	• PROVIDE REQMTS FOR PARALLEL TIME STAGE PROCESSING FOR HIGH LAUNCH RATES • EACH SEPARATE STAGE UNDERGOES REFURB. & REVERIFICATION IN ITS OWN PROCESSING STATION
9. GSE & SUPPORT SYSTEMS OPERATIONS	• DESIGN GSE TO SATISFY MAXIMUM NUMBER OF CHECKOUT REQMTS FOR EACH OF THE SYSTEMS • GSE DESIGN INCLUDES AUTOMATED C/O OF SYSTEMS AT TIME OF ACTIVATION/DEACTIVATION
10. GROUND CHECKOUT AND SYSTEMS TESTS	• REVERIFYING FLIGHT SYST. REDUCED BY USE OF ON-BOARD FAULT ISOLATION & AUTOMATED DESIGN • DESIGN FOR REDUCED REDUNDANT C/O WHICH REQUIRES TIME & CONTRIBUTES TO HARDWARE WEAROUT
11. STAGES TRANSFERRED TO LAUNCH PAD	• VEHICLE DESIGN PROVIDES FOR TOWING TO PAD AND ROTATING TO VERTICAL ON PAD
12. STAGES ATTACHED TO LAUNCH PAD	• MINIMIZE & AUTOMATE INTERFACE CABLES, FLUID LINES, & GSE CONNECTIONS • MINIMIZE ALIGNMENT CHECKS, PREP OF MATING INTERFACES & VEHICLE-TO-PAD INTERFACES
13. 1ST & 2ND STAGES MATING & INTEGRATION	• SIMPLIFY INTER-STAGE INTERFACE MATING REQMTS, ALIGNMENT, & INTERFACE VERIFICATION • DESIGN MAIN ENGINE HEAT SHIELDS & TPS CLOSEOUT FOR EASE OF MAINT. & INSTALLATION
14. LAUNCH READINESS VERIFICATION TEST	• TESTS DESIGNED FOR AUTOMATIC CHECKING & SELF-TESTING OF LAUNCH-CRITICAL FUNCTIONS • PROVIDE FOR END-TO-END FUNCTIONAL VERIFICATION & LAUNCH READINESS STATUS CHECK
15. SERVICING/ON-PAD PROPELLANT LOADING	• DESIGN MOST ON-BOARD SYST TO USE SAME FUELS TO PROVIDE FOR PARALLEL FUELING OF SYST. • DESIGN HAZARDOUS SERVICING FOR STAGGERED-START SIMULTANEOUS FLOW TO MINIMIZE PAD CLEAR.
16. FINAL LAUNCH OPERATIONS AND COUNTDOWN	• SIMPLIFY REQUIRED STATUS VERIFICATIONS, ALIGNMENTS, AND PRESSURE INTEGRITY CHECKS

is already available, but the relative merits of each must be determined. TPS inspection and maintenance is the key operations driver in the current STS program. In addition, the projected, life of the STS is limited by the structural thermal cycling environments.

In order to meet the postulated operational requirements of the SPS transportation system, the cryogenic tanks of both the booster and the orbiter must be designed such that they require little or no inspection outside of normal maintenance cycles. Similar requirements are placed on the tank insulation also. The tanks of the present point design are identified as being non-integral structurally with the tank insulation system, permitting relatively easy inspection when required, but not allowing the buildup of icing on the external surfaces nor any cryopumping. A number of candidate insulation systems has been identified in past studies and, as in the case of the TPS, each has its own merits but none can completely satisfy SPS cryogenic tank insulation requirements. Materials and systems design technology must be pursued before any firm decision can be made on systems selection.

In order to minimize vehicle turnaround requirements and cost, all vehicle systems will require minimum inspection, maintenance, and replacement. This is especially true of the liquid rocket engines which are second only to the thermal protection system in turnaround operations requirements of the STS. Improvements in materials and design technology improvement in the critical areas of turbine pumps and seals, regulator valves, and precombustion chamber components are required to satisfy nominal turnaround operations and cost.

A great dependence must be placed upon on-board monitoring and fault detection/isolation systems in order to preclude the requirement for ground interfacing and checkout requirements. All previous ground and flight performance data must be computer analyzed to determine performance trend data indicative of potential impending failures. Methods of implementation and types of diagnostic monitoring equipment must be evaluated and defined.

All cargo must arrive at the launch site in a pre-palletized configuration in order to minimize handling. Cargo manifests will be comp ter controlled with automated cargo handling and transfer.

Communications between the launch vehicle and ground stations will be restricted such that the launch vehicle is essentially capable of autonomous operation other than launch and landing clearances.

In order to define SPS operational requirements, an in-depth analysis of the STS turnaround timeline assessment was conducted to determine the cost/time drivers, and to identify those operations which might be deleted by imposing new operations requirements on the SPS HLLV. Table 6.1-3 and Figures 6.1-1 and 6.1-2 are taken from the STS "STAR" report.

Table 6.1-3 is representative of the detailed analyses required to establish credible STS turnaround timelines and requirements.

Figure 6.1-1 depicts the orbiter flow from touchdown to launch readiness. The most optimistic turnaround assessment indicates a turnaround processing requirement of 205 hours (~8.5 days).

Table 6.1-3. Level III Allocations/Assessment Deltas

Alloc.	Assessment Previous Report	Assessment Present Report	Alloc. Assmt. Delta	Assmt. Increase Impact	Item Description	SPO	Hardware Maturity Status
1.0	1.0	1.0	0.0	-	LANDING AREA	-	-
1.0	1.0	1.0	0.0	-	Postlanding Operations	-	-
0.5	0.5	0.5	0.0	-	Provide ECLSS Coolant and Orbiter Purge	-	-
1.0	1.0	1.0	0.0	-	Crew Exchange	-	-
96.0 87.5	206.5 134.0	205.0 126.5	109.0 39.0	(-1.5) (-7.5S)	ORBITER PROCESSING FACILITY(SL) ORBITER PROCESSING FACILITY(US)	-	-
0.5	1.0	1.0	0.5	-	Tow to OPF	L&L	Fabrication complete
2.5	2.5	2.5	0.0	-	Transfer to Facility Services	-	-
3.0	3.0	3.0	0.0	-	Jack and Level	-	-
0.5	2.0	2.0	1.5	-	Position Orbiter Access Platforms. Task relocated ahead of purge and dry SSME operations and becomes totally serial.	L&L	Basic design complete
2.0	4.0	16.0	14.0	(-3.0S)	Purge and Dry SSME - PRCBD 4123 resulted in a longer (12-hr minimum) purge. The purge will be vented, via yet to be defined GSE, external to the OPF. This allows the purge to be a nonhazardous, parallel operation. The task includes 3 hrs. of preps, 12 hrs. for the purge and 1 hr. to obtain dew point moisture samples and secure.		
-	0.5	0.5	0.5	-	Clear Nonessential Personnel	L&L	N/A
-	1.5	1.5	1.5	-	APS/FRCS Pod Safing	ORB	-
3.0	4.0	4.0	1.0	-	Vent, Drain, & Purge PRSD	ORB	-
0.5	2.0	2.0	1.5	-	Vent ECLSS GO_2/GN_2	L&L	-
4.0	6.0	6.0	2.0	-	Prep and Service APU	ORB	-
-	0.5	0.5	0.5	-	Open Noncontrolled Areas	L&L	N/A

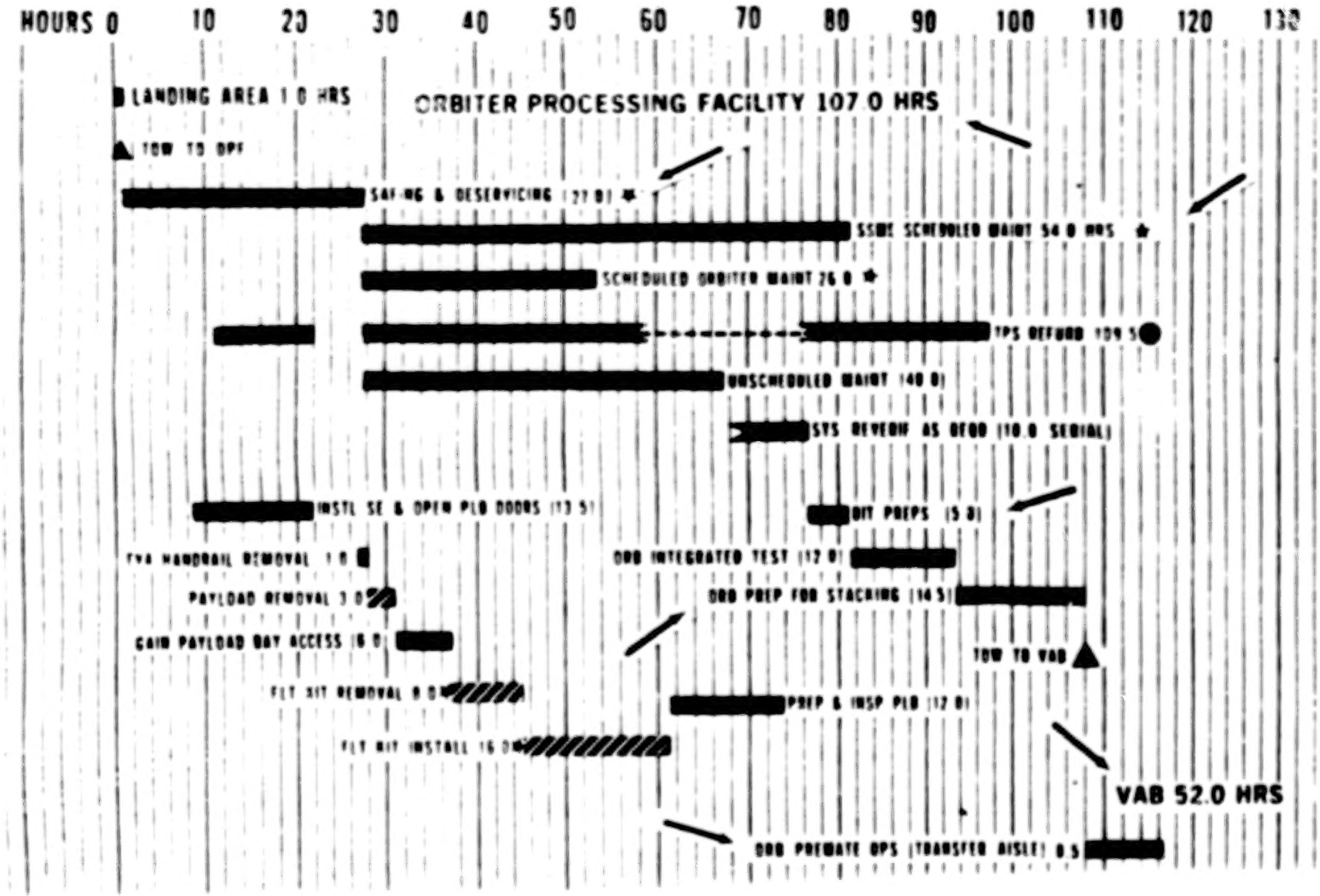


Figure 6.1-1. STS Turnaround Timeline Assessment

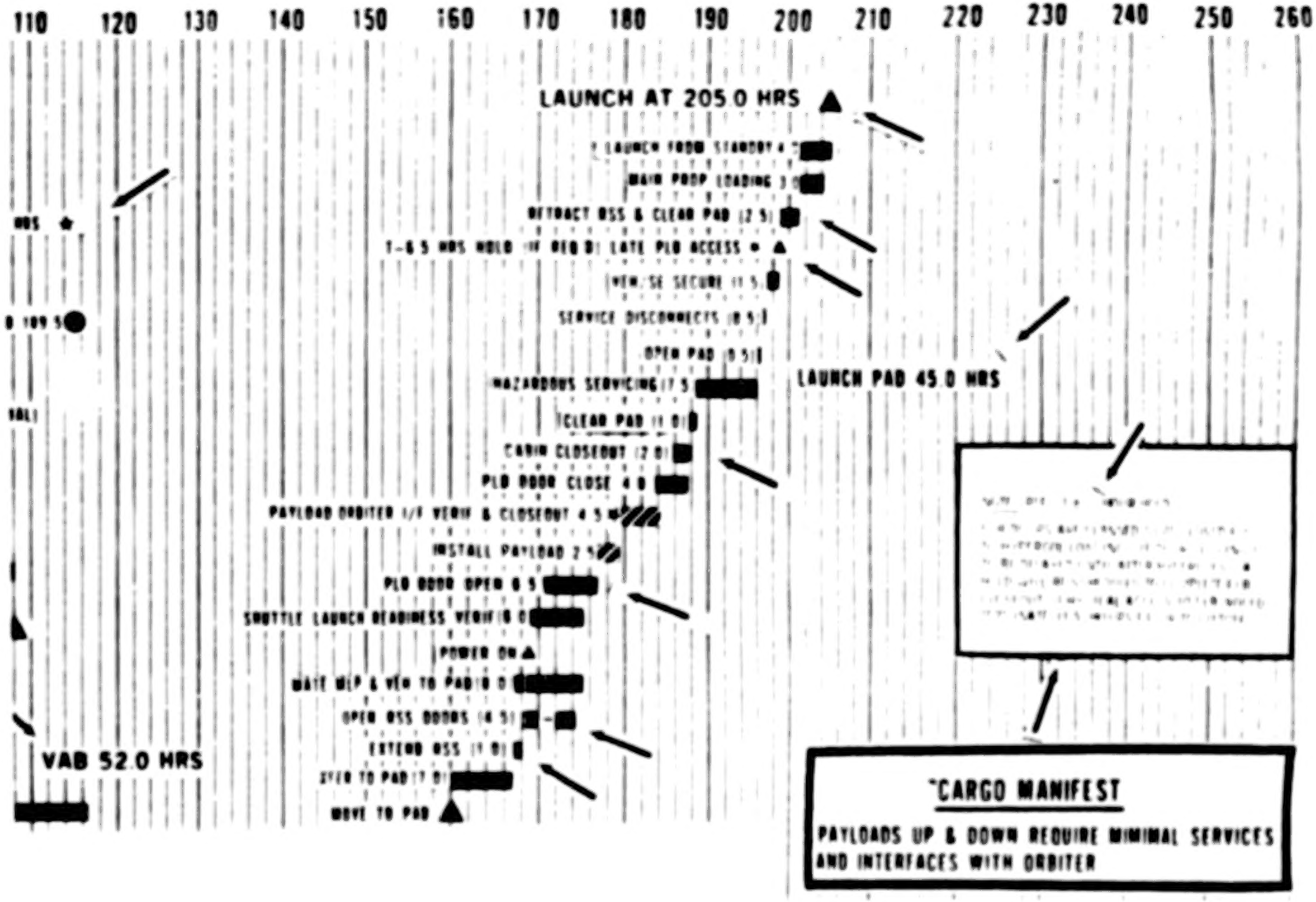


Figure 6.1-1. STS Turnaround Timeline Assessment (Cont)

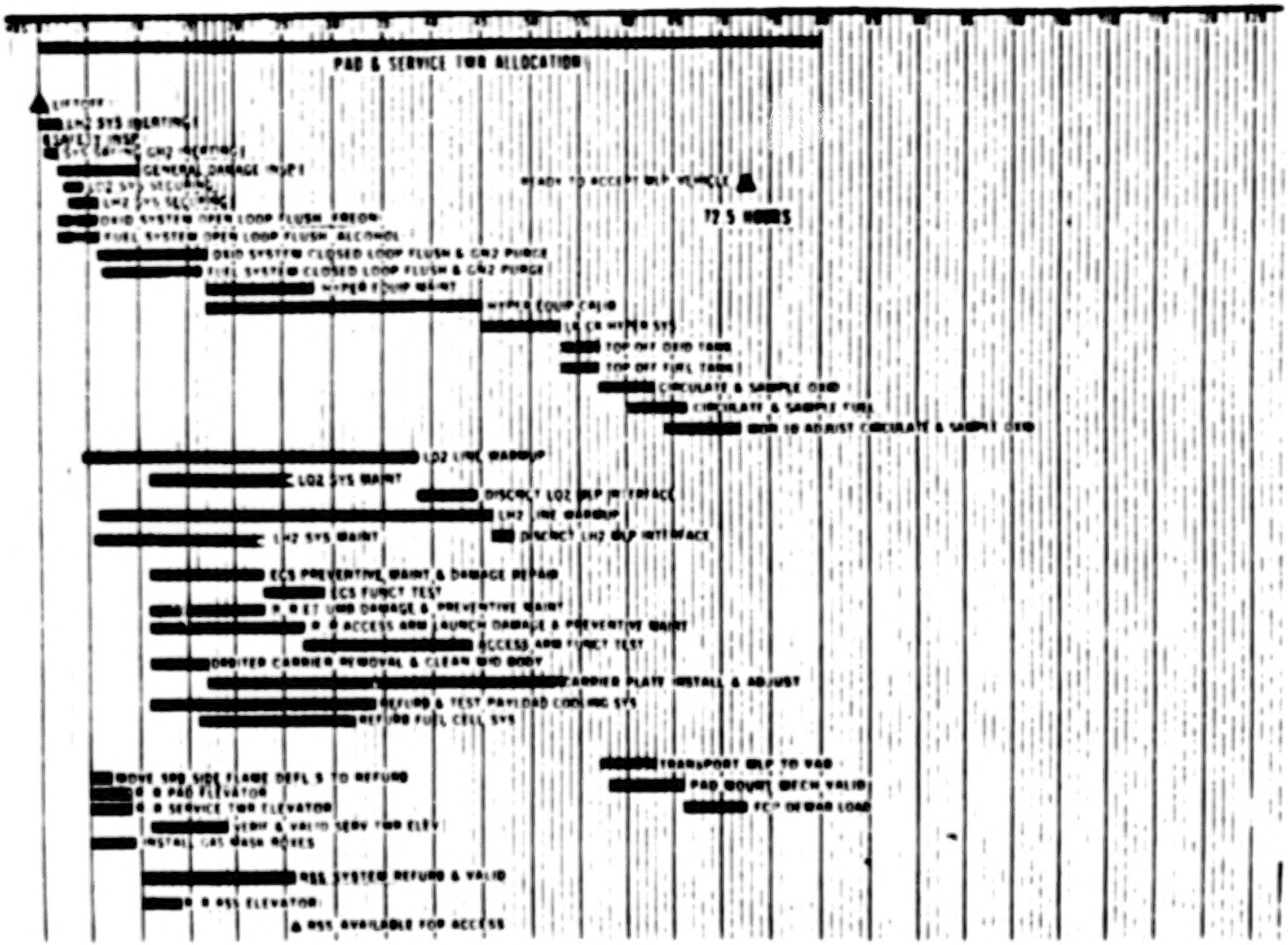


Figure 6.1-2. STS Launch Pad Turnaround Assessment

Figure 6.1-2. STS Launch Pad Turnaround Assessment (Cont)

Figure 6.1-2 defines the launch pad turnaround requirements. The turnaround schedule is based on minimal launch pad refurbishment after each flight. As indicated, pad turnaround from liftoff to next launch is 72 hours (3 days).

Figure 6.1-3 is self-explanatory. It is noted that the original turnaround allocation for STS was 160 hours (~7 days). The previous timelines presented show a schedule of 205 hours, whereas the current (end of 1978) assessment is indicating 235 hours. In the same manner that "growth margins" have been included in systems design, consideration of potential schedule growth must also be considered in HLLV turnaround estimates.

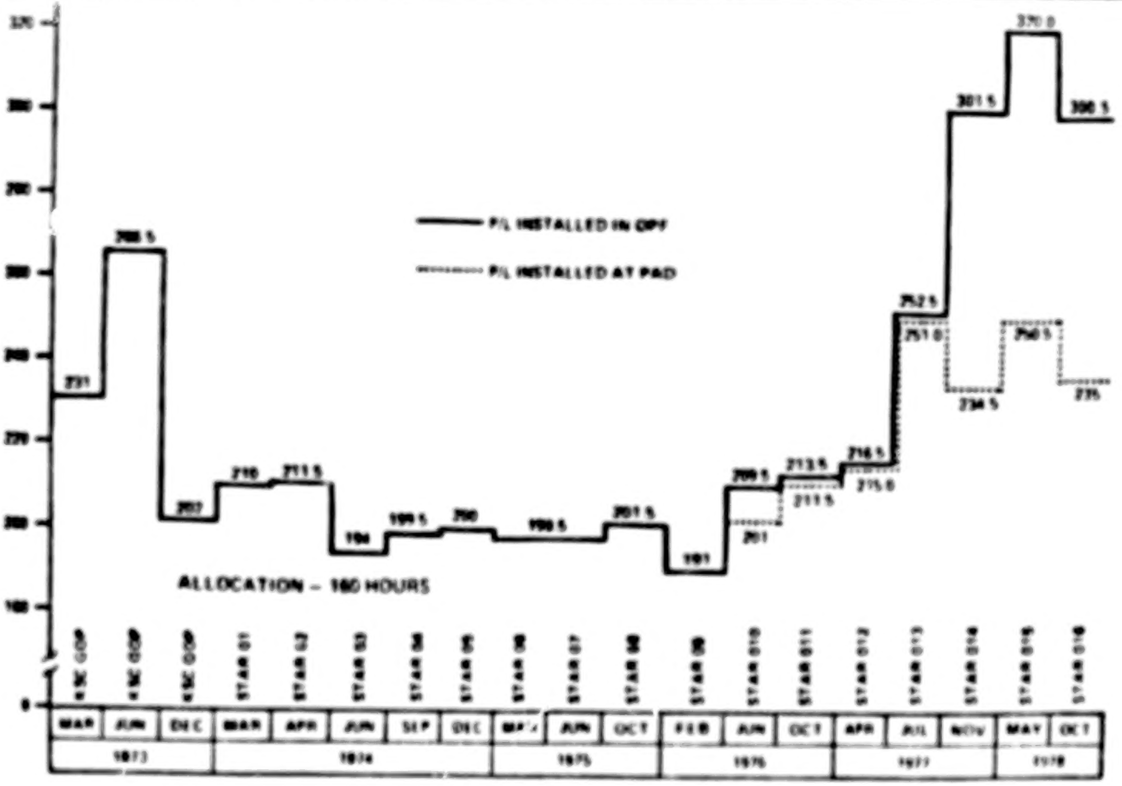


Figure 6.1-3. Shuttle Turnaround Assessment History

A most optimistic assessment of potential turnaround timeline requirements for the SPS-HLLV indicates a minimum orbiter turnaround capability of two days (excluding mission time) and a pad turnaround of 1-1/2 days.

In order to meet these requirements, it is assumed that the HLLV can be fueled in the same period of time as the STS. The current STS fuel flow rates are approximately 1250 GPM with a potentaal of 5000 GPM. An order of magnitude increase will be required for the HLLV.

It is also noted that vehicle maintenance and checkout operations must be conducted in parallel with pad operations (excluding propellant servicing).

If STS experience is applied to these estimates and mission time included, a credible turnaround estimate for the orbiter would be 4 days with a pad turnaround time of 3 days; for a total HLLV turnaround of 7 days.

In Summary, HLLV operations are a prime cost driver and a key to SPS Construction cost credibility. Regardless of configuration or concept selected to satisfy HLLV requirements, there are several key technology development requirements common to all. The need for thermal protection and insulation systems which are capable of repeated use without the need for extensive maintenance and inspection is a must.

Liquid rocket engines with the capability of extended life with little or no maintenance—again regardless of size or propellant type—will be a major driver in reducing operations costs.

An on-board, self-monitoring/checkout system will alleviate the need for ground interfaces and expensive ground checkout facilities and time-consuming subsystems flight-readiness determination.

Ground servicing and handling procedures must closely parallel those employed in air transport systems. A thorough analysis of ground systems requirements and definition must be pursued.

6.2 ORBITAL OPERATIONS DEFINITION

As previously stated, a LEO staging base will be required for crew/cargo transfer and orbital vehicles maintenance. The HLLV will rendezvous only with the LEO base (i.e., docking not required). Cargo will then be transferred from the HLLV to the EOTV by LEO based on-orbit transfer vehicles. Down payload, as required, will be transferred to the HLLV. A maximum stay time on orbit for the HLLV should not exceed twelve hours.

The PLV may rendezvous or dock with the LEO base in order to effect crew transfer. The crew module will be removed from the PLV cargo bay and mated to a single stage POTV element for immediate transfer to GEO. Crews returning to earth will have already boarded a crew module, which will then be loaded into the PLV cargo bay. The maximum stay time for the PLV in LEO will be twelve hours.

LEO base maintenance of orbital vehicles will be primarily restricted to component (LRU) replacements on the EOTV, POTV and on-orbit mobility units; and the propellant servicing requirements of the POTV and OOMU. (EOTV propellant tanks will be transferred directly from the HLLV to the EOTV).

The EOTV, POTV and OOMU GEO operations will be essentially the same as those conducted in LEO. Transportation system maintenance provisions in GEO will also be the same as those in LEO.

The POTV shares common technology requirements with the HLLV (i.e., propellant tank insulation, engine component life, self-monitoring/diagnostic equipment, etc.), and can benefit from those technology programs implemented for the HLLV. A unique technology requirement of the POTV is in the area of

orbital maintenance. Continuation of on-going orbital propellant transfer technology programs may satisfy this requirement. Engine overhaul/replacement should be an earth-based operation due to the potential complexity and limited advantage of performing that function at the orbital bases. Emergency repairs only should be pursued.

The EOTV shares common operations technology features with the SPS (i.e., the EOTV utilizes the same power source and design features as the SPS). The unique operations requirement of the EOTV is that it must be capable of repeatedly transitioning the Van Allen radiation belt with minimum degradation. In addition, the ion thrusters employed for the EOTV are of a higher current density than the SPS (i.e., to achieve higher thrust) and must therefore be capable of periodic screen grid replacement at the orbital base(s). The EOTV propellant distribution system is designed to permit fueled tank replacement in lieu of propellant transfer from an orbital propellant depot. This eliminates the need for additional orbital tank farms, minimizes propellant boil-off and transfer losses and permits transport of lower density payloads with the high density loaded argon tanks.

As in the case of the HLLV, the orbital systems design are oriented towards minimal handling and manpower requirements in order to reduce the size and manpower complement of the orbital bases.

7.0 COST AND PROGRAMMATICS

Cost and programmatic data (i.e., schedules, technology requirements, etc.) were developed for the several SPS options evaluated. The cost and scheduling data are included in Volume VI (Cost and Programmatics) and the technology advancement requirements and task plans are included in Volume V (Systems Engineering/Integration Research and Technology) and, therefore will not be repeated in this document. Included in this section are the detailed traffic models and comparative assessment of vehicle fleet and flight requirements for the several SPS concepts evaluated. The specific SPS concepts for which traffic models were developed included:

- The updated GaAs reference concept (Exhibit C)
- The GaAs reference concept with magnetron antenna
- The GaAs reference concept with dual solid-state antennas
- A dual-sandwich concept with standard GaAs cells
- A dual-sandwich concept with multi-bandgap cells

(Please refer to Volume II, Systems/Subsystems Analyses for a complete satellite description.)

7.1 SATELLITE ANNUAL MAINTENANCE MASS

In order to develop traffic models for the alternate SPS concepts, it was necessary to establish the various maintenance mass requirements since they are a significant contributor to overall transportation requirements. The annual maintenance mass for the alternate concepts are presented in Table 7.1-1. The primary difference in maintenance mass requirements is in the area of antenna maintenance.

Table 7.1-1. Satellite Annual Maintenance Requirements (10^6 kg)

COMPONENT	REF. GaAs	GaAs (MAG ANT.)	GaAs (DUAL SS ANT.)	DUAL SANDWICH	DUAL SANDWICH (MBG)
STRUCTURES	0.003	0.002	0.003	0.004	0.003
MECHANISMS	0.001	0.001	0.002	-	-
CONCENTRATORS	0.001	0.001	0.001	0.002	0.002
SOLAR PANELS	0.007	0.007	0.008	-	-
POWER DIST./CONTROL	0.110	0.137	0.121	-	-
MAINTENANCE PROV.	-	-	0.001	0.001	-
INFORMATION MGMT/CONT.	0.001	0.001	0.003	0.001	0.001
ACS	0.002	0.002	0.002	0.002	0.002
ANTENNA SUBARRAY	0.353	-	-	-	-
ANT. CONT. ELEC.	0.002	0.002	0.003	0.003	0.002
SUBTOTAL	0.480	0.153	0.144	0.013	0.012
GROWTH (25%)	0.120	0.038	0.036	0.003	0.003
PROP. & TANKS	0.066	0.066	0.066	0.180	0.180
TOTAL	0.666	0.257	0.246	0.196	0.195

7.2 COMPARATIVE TRAFFIC MODELS

Table 7.2-1 presents a comparison of vehicle flight requirements for the construction of the precursor or pilot-plant satellite. During this early program phase, the STS and its derivatives are used exclusively for earth to LEO transportation. The pilot plant vehicles are constructed in LEO.

Table 7.2-1. Comparative Flight Requirements - Precursor Satellite

	VEHICLE FLIGHT REQUIREMENTS				
	STS-PLV	STS-CARGO	STS-GROWTH	STS-HLLV	EOTV-EQUIV
REFERENCE GaAs SATELLITE					
KLYSTRON ANTENNA	6	113	72	63	
MAGNETRON ANTENNA	6	101	72	63	
DUAL SOLID-STATE ANT	6	122	72	63	
NEW SATELLITE CONCEPT					
STANDARD CELLS	12	241	72	168	1
MBG CELLS	12	206	72	168	1

The greater flight requirements for the new satellite concept, precursor or pilot plant, are due to the fact that the "new SPS concept" pilot plant is structurally a complete satellite, whereas the reference concept SPS utilizes an EOTV derivative for pilot plant demonstration. Being an EOTV derivative, the reference configurations are self-transportable from LEO to GEO, whereas, an equivalent EOTV propulsive element is required to transport the unit to GEO.

Table 7.2-2 summarizes the flight and fleet requirements for the construction of the first satellite, theoretical first unit (TFU).

Table 7.2-2. Comparative Flight/Fleet Requirements - TFU

	VEHICLE FLIGHT/FLEET REQUIREMENTS			
SATELLITE CONFIGURATION	HLLV	POTV	EOTV	IOTV
REFERENCE GaAs SATELLITE				
KLYSTRON ANTENNA	245/5	40/4	8/6	463/4
MAGNETRON ANTENNA	217/5	40/4	7/6	411/4
DUAL SOLID-STATE ANT.	290/5	40/4	9/6	594/4
NEW SATELLITE CONCEPT				
STANDARD CELLS	170/5	60/5	4/4	337/4
MBG CELLS	146/5	60/5	3/3	293/4

The fewer flights required for the new SPS concept TFU are due to the reduced weight (and power) of the satellite. The magnetron antenna version of the reference SPS concept shows a significant improvement over the reference concept because of the reduced mass to orbit requirement.

The comparative flight and fleet requirements for the total SPS construction and operation program of 60 years are presented in Table 7.2-3.

Table 7.2-3. Comparative Flight/Fleet Requirements - 60-Year Program

SATELLITE CONFIGURATION	NUMBER OF SATELLITES	VEHICLE FLIGHT/FLEET REQUIREMENTS			
		HLLV	POTV	EOTV	IOTV
REFERENCE GaAs SATELLITE					
KLYSTRON ANTENNA	60	13,994/47	1544/15	396/20	27,662/139
MAGNETRON ANTENNA	54	11,288/38	1512/15	318/16	22,332/112
DUAL SOLID-STATE ANTENNA	58	16,298/54	1533/15	463/23	31,711/159
NEW SATELLITE CONCEPT					
STANDARD CELLS	125	19,953/67	2372/24	564/28	39,152/196
MBG CELLS	90	13,189/44	1773/18	371/19	26,044/130

The total program transportation requirements for the magnetron antenna concept are definitely most favorable from the aspect of transportation costs. The new SPS concept suffers from the requirement for a greater number (lower power) of satellites.

7.3 DETAILED TRAFFIC MODELS

The comparative traffic models presented in 7.2 were developed from the detailed models presented herein. Tables 7.3-2 through 7.3-16 summarize the flight and fleet requirements for the three program phases: precursor or "pilot plant" construction, theoretical first unit production, and the total 60-year construction and operations program for the five SPS options. Table 7.3-1 presents the Exhibit "C" reference concept for comparison with the updated reference. The major reductions in flight/fleet requirements (and presumably cost) are the result of changes in the klystron maintenance concept and the utilization of the HLLV for personnel transport to LEO in lieu of the STS.

A 10% packaging factor is included in all mass delivery requirements and the HLLV and EOTV have a 10% return payload capability.

The fleet requirements for the TFU construction are the minimum number required to meet traffic model demands. The operational fleet requirements are based on a useful flight life of 300 flights/HLLV, 100 flights/POTV, 20 flights/EOTV, and 200 flights/IOTV.

Table 7.3-1. GaAs Exhibit C Reference SPS Concept—Total Program Transportation Requirements, 60-Year Program (60 Satellites)

	MASS x 10⁶ KG		VEHICLE FLIGHTS					
	LEO	GEO	PLV	HLLV	POTV	EOTV	IOTV LEO	IOTV GEO
SATELLITE								
CONSTRUCTION	2197.8	2197.8	1340	9682	1220	425.1	9682	9682
OPERATIONS & MAINTENANCE	1803.0	1803.0	3694	7943	3660	348.7	7943	7943
CREW CONSUMABLES								
CONSTRUCTION	31.5	28.7	-	139	-	5.6	139	126
OPERATIONS & MAINTENANCE	86.8	86.0	-	382	-	16.6	382	379
POTV PROPELLANTS								
CONSTRUCTION	82.7	41.4	-	364	-	8.0	364	182
OPERATIONS & MAINTENANCE	267.8	133.8	-	1180	-	25.9	1180	589
EOTV CONSTRUCTION								
CONSTRUCTION	28.2	24.2	-	124	-	4.7	124	107
OPERATIONS & MAINTENANCE	22.2	19.0	-	98	-	3.7	98	84
EOTV PROPELLANTS								
CONSTRUCTION	340.3	2.0	-	1499	-	0.4	1499	9
OPERATIONS & MAINTENANCE	304.0	-	-	1339	-	-	1339	-
IOTV PROPELLANTS								
CONSTRUCTION	7.2	3.3	-	32	-	0.6	32	15
OPERATIONS & MAINTENANCE	6.6	3.0	-	29	-	0.6	29	13
SUMMARY								
CONSTRUCTION	2687.7	2297.4	1340	11840	1220	444	11840	10121
OPERATIONS & MAINTENANCE	2490.4	2044.8	3694	10971	3660	396	10971	9008
TOTAL	5178.1	4342.2	5034	22811	4880	840	22811	19129
VEHICLE FLEET								
CONSTRUCTION	-	-	14	39	12	22	110	
OPERATIONS & MAINTENANCE	-	-	37	37	37	20	100	
TOTAL	-	-	51	76	49	42	210	

Table 7.3-2. GaAs Reference SPS Concept—Total Transportation Requirements, 60-Year Program (60 Satellites)

	MASS × 10⁶ kg		VEHICLE FLIGHTS					
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	IOTV LEO	IOTV GEO
SATELLITE								
CONSTRUCTION	2087.7	2087.7	111	9,197	1220	306.4	10,741	9,197
OPS & MAINT.	492.2	492.2	34	2,168	324	72.7	2,560	2,168
CREW CONSUMABLES								
CONSTRUCTION	29.9	28.7		132		4.2	132	126
OPS & MAINT.	9.2	7.6		41		1.1	41	34
POTV PROPELLANTS								
CONSTRUCTION	87.9	44.0		387		6.5	387	194
OPS & MAINT.	23.3	11.7		103		1.7	103	52
EOTV CONSTRUCTION								
CONSTRUCTION	19.9	12.4		88		1.8	88	55
OPS & MAINT.	5.0	5.0		22		0.7	22	22
EOTV PROPELLANTS								
CONSTRUCTION	306.0	1.9		1,348		0.3	1,348	8
OPS & MAINT.	73.0	0.8		322		0.1	322	4
IOTV PROPELLANTS								
CONSTRUCTION	7.4	3.2		33		0.5	33	14
OPS & MAINT.	1.8	0.8		8		0.1	8	3
SUMMARY								
CONSTRUCTION	2538.8	2177.9	111	11,185	1220	320	12,729	9,594
OPS & MAINT.	604.5	518.1	34	2,664	324	76	3,056	2,283
TOTAL	3143.3	2696.0	145	13,849	1544	396	15,785	11,877
VEHICLE FLEET								
CONSTRUCTION	-	-	-	38	12	16	112	
OPS & MAINT.	-	-	-	9	3	4	27	
TOTAL	-	-	-	47	15	20	139	

Table 7.3-3. GaAs Reference SPS Concept (Magnetron Antenna)—
Total Transportation Requirements, 60-Year Program
(54 Satellites)

	MASS – 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE								
CONSTRUCTION	1589.5	1589.5	111	7,002	1220	233.3	8,276	7,002
OPS & MAINT.	471.5	471.5	32	2,077	292	69.2	2,437	2,077
CREW CONSUMABLES								
CONSTRUCTION	29.9	28.7		132		4.2	132	126
OPS & MAINT.	7.6	6.9		34		1.0	34	30
POTV PROPELLANTS								
CONSTRUCTION	87.9	44.0		387		6.5	387	194
OPS & MAINT.	10.5	5.3		46		0.8	46	23
EOTV CONSTRUCTION								
CONSTRUCTION	14.9	7.5		66		1.1	66	33
OPS & MAINT.	5.0	5.0		22		0.7	22	22
EOTV PROPELLANTS								
CONSTRUCTION	234.9	1.1		1,035		0.2	1,035	5
OPS & MAINT.	69.2	0.8		305		0.1	305	4
IOTV PROPELLANTS								
CONSTRUCTION	6.8	2.9		30		0.4	30	13
OPS & MAINT.	2.0	0.9		9		0.1	9	4
SUMMARY								
CONSTRUCTION	1963.9	1673.7	111	8,652	1220	246	9,926	7,373
OPS & MAINT.	565.8	490.4	32	2,493	292	72	2,853	2,160
TOTAL	2529.7	2164.1	143	11,145	1512	318	12,779	9,553
VEHICLE FLEET								
CONSTRUCTION	-	-	-	30	12	12	87	
OPS & MAINT.	-	-	-	8	3	4	25	
TOTAL	-	-	-	38	15	16	112	

Table 7.3-4. GaAs Reference Concept (Dual Solid-State Antenna)—
Total Transportation Requirements, 60-Year Program
(58 Satellites)

	MASS – 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE								
CONSTRUCTION	2550.1	2550.1	111	11,234	1220	374.2	12,508	11,234
OPS & MAINT.	485.3	485.3	33	2,138	313	71.2	2,519	2,138
CREW CONSUMABLES								
CONSTRUCTION	29.9	28.7		132		4.2	132	126
OPS & MAINT.	9.0	7.4		39		1.1	39	32
POTV PROPELLANTS								
CONSTRUCTION	87.9	44.0		387		6.5	387	194
OPS & MAINT.	22.6	11.3		99		1.7	99	50
EOTV CONSTRUCTION								
CONSTRUCTION	23.6	16.2		104		2.4	104	71
OPS & MAINT.	5.0	5.0		22		0.7	22	22
EOTV PROPELLANTS								
CONSTRUCTION	371.2	2.5		1,535		0.4	1,635	11
OPS & MAINT.	72.1	0.8		318		0.1	318	4
IOTV PROPELLANTS								
CONSTRUCTION	8.7	3.9		38		0.6	38	17
OPS & MAINT.	1.7	0.7		7		0.1	7	3
SUMMARY								
CONSTRUCTION	3071.4	2645.4	111	13,530	1220	388	14,804	11,653
OPS & MAINT.	595.7	510.5	33	2,624	313	75	3,005	2,249
TOTAL	3667.1	3155.9	144	16,154	1533	463	17,809	13,902
VEHICLE FLEET								
CONSTRUCTION	-	-	-	46	12	19	132	
OPS & MAINT.	-	-	-	8	3	4	27	
TOTAL	-	-		54	15	23	159	

Table 7.3-5. GaAs Dual Sandwich SPS Concept—
Total Transportation Requirements,
60-Year Program (125 Satellites)

	MASS - 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE								
CONSTRUCTION	2822.7	2822.7	167	12,495	18[illegible]	414.3	14,349	12,495
OPS & MAINT	839.8	839.8	51	3,699	5	123.2	4,281	3,699
CREW CONSUMABLES								
CONSTRUCTION	45.0	43.7		198		6.4	198	193
OPS & MAINT	13.7	12.0		60		1.8	60	53
POTV PROPELLANTS								
CONSTRUCTION	134.0	67.0		590		9.8	590	295
OPS & MAINT	36.9	18.5		163		2.7	163	81
EOTV CONSTRUCTION								
CONSTRUCTION	27.3	19.9		120		2.9	120	88
OPS & MAINT	7.5	7.5		33		1.1	33	33
EOTV PROPELLANTS								
CONSTRUCTION	416.5	3.1		1,835		0.5	1,835	14
OPS & MAINT	123.7	1.1		545		0.2	545	5
IOTV PROPELLANTS								
CONSTRUCTION	10.0	4.3		44		0.6	44	19
OPS & MAINT	3.0	1.3		13		0.2	13	6
SUMMARY								
CONSTRUCTION	3455.6	2960.7	167	15,222	1860	435	17,136	13,044
OPS & MAINT	1024.6	880.2	51	4,513	512	129	5,095	3,877
TOTAL	4480.2	3840.9	218	19,735	2372	564	22,231	16,921
VEHICLE FLEET								
CONSTRUCTION	-	-	-	51	19	22	151	
OPS & MAINT	-	-	-	16	5	6	45	
TOTAL	-	-	-	67	24	28	196	

Table 7.3-6. Dual Sandwich SPS Concept (MBG Cells)—
Total Transportation Requirements
60-Year Program (98 Satellites)

	MASS - 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE								
CONSTRUCTION	1766.4	1766.4	132	7,782	1458	259.2	9,294	7,782
OPS & MAINT	630.6	630.6	34	2,778	315	92.6	3,163	2,778
CREW CONSUMABLES								
CONSTRUCTION	35.5	34.3		157		5.0	157	151
OPS & MAINT	9.1	7.4		40		1.1	40	33
POTV PROPELLANTS								
CONSTRUCTION	105.1	52.5		463		7.7	463	231
OPS & MAINT	22.7	11.4		100		1.7	100	50
EOTV CONSTRUCTION								
CONSTRUCTION	17.4	9.9		77		1.5	77	44
OPS & MAINT.	6.2	6.2		27		0.9	27	27
EOTV PROPELLANTS								
CONSTRUCTION	261.9	1.5		1,154		0.2	1,154	7
OPS & MAINT.	92.2	1.0		406		0.2	406	4
IOTV PROPELLANTS								
CONSTRUCTION	6.4	2.7		28		0.4	28	12
OPS & MAINT.	2.6	1.1		11		0.2	11	5
SUMMARY								
CONSTRUCTION	2192.7	1867.3	132	9,661	1458	274	11,173	8,227
OPS & MAINT.	763.4	657.7	34	3,362	315	97	3,747	2,897
TOTAL	2956.1	2525.0	166	13,023	1773	371	14,920	11,124
VEHICLE FLEET								
CONSTRUCTION	-	-	-	33	15	14	97	
OPS & MAINT.	-	-	-	11	3	5	33	
TOTAL	-	-	-	44	18	19	130	

Table 7.3-7. GaAs Reference SPS Concept—
Precursor Transportation Requirements

	MASS × 10^6 kg	VECHICLE FLIGHTS			
		STS (PLV)	STS (CARGO)	STS-GROWTH (PLV)	STS-HLLV (CARGO)
PRECURSOR	2.019	6	79	-	-
LEO BASE	5 MODULES	-	-	-	5
SCB	5.300	-	-	72	58
PROPELLANT	0.864	-	34	-	-
TOTAL	-	6	113	72	63

Table 7.3-8. GaAs Reference SPS Concept—
TFU Transportation Requirements

	MASS × 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE CONSTR. & MAINT	34.8	34.8	5.4	153.3	40	5.1	215	153
CREW CONSUMABLES	1.5	0.1		6.6		-	7	-
POTV PROPELLANTS	2.9	1.4		12.7		0.2	13	6
EOTV CONSTRUCTION & MAINT	7.5	-		32.8		-	33	-
EOTV PROPELLANTS	7.6	-		33.5		-	34	-
IOTV PROPELLANTS	0.2	0.1		0.6		-	1	1
SCB TO GEO	-	-		-		2	-	-
TOTAL	54.5	36.4	5	240	40	8	303	160
FLEET	-	-	-	5	4	6	2	2

Table 7.3-9. Reference SPS Concept (Magnetron Antenna)—
Precursor Transportation Requirements

	MASS × 10^6 kg	VECHICLE FLIGHTS			
		STS (PLV)	STS (CARGO)	STS-GROWTH (PLV)	STS-HLLV (CARGO)
PRECURSOR	1.746	6	67	-	-
LEO BASE	5 MODULES	-	-	-	5
SCB	5.300	-	-	72	58
PROPELLANT	0.864	-	34	-	-
TOTAL	-	6	101	72	63

Table 7.3-10. GaAs Reference SPS Concept (Magnetron Antenna—TFU Transportation Requirements

	MASS - 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE CONSTR. & MAINT.	44.0	44.0	5.4	193.[illegible]	40	6.5	256	194
CREW CONSUMABLES	1.5	0.1		6.6		-	7	-
POTV PROPELLANTS	2.9	1.4		12.[illegible]		0.2	13	6
EOTV CONSTRUCTION & MAINT.	7.5	-		32.8		-	33	-
EOTV PROPELLANTS	8.6	-		37.7		-	38	-
IOTV PROPELLANTS	0.2	0.1		0.9		-	1	1
SCB TO GEO	-	-		-		2	-	-
TOTAL	64.7	45.6	5	285	40	9	348	201
FLEET	-	-	-	5	4	6	2	2

Table 7.3-11. Reference SPS Concept (SS Antenna)—Precursor Transportation Requirements

		VEHICLE FLIGHTS			
	MASS - 10^6 kg	STS (PLV)	STS (CARGO)	STS-GROWTH (PLV)	STS-HLLV (CARGO)
PRECURSOR	2.320	6	88	-	-
LEO BASE	5 MODULES	-	-	-	-
SCB	5.300	-		72	5
PROPELLANT	0.864	-	34	-	58
TOTAL	-	6	122	72	63

Table 7.3-12. Reference SPS Concept (SS Antenna)—TFU Transportation Requirements

	MASS - 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE CONSTR. & MAINT.	29.4	29.4	5.4	129.7	40	4.3	192	130
CREW CONSUMABLES	1.5	0.1		6.6		-	7	-
POTV PROPELLANTS	2.9	1.4		12.7		0.2	13	6
EOTV CONSTRUCTION & MAINT.	7.5	-		32.8		-	33	-
EOTV PROPELLANTS	6.7	-		29.[illegible]		-	29	-
IOTV PROPELLANTS	0.1	-		0.6		-	1	-
SCB TO GEO	-	-		-		2	-	-
TOTAL	48.1	30.9	5	212	40	7	275	136
FLEET	-	-	-	5	4	6	2	2

Table 7.3-13. Dual Sandwich SPS Concept—
Precursor Transportation Requirements

	MASS—10^6 kg	VEHICLE FLIGHTS				
		STS		STS-GROWTH (PLV)	STS-HLLV (CARGO)	EOTV EQUIV GEO R'FLR
		PLV	CARGO			
PRECURSOR	4.67	12	171	-	-	1
LEO BASE	5 MODULES	-	-	-	5	-
SCB	14.82	-	-	72	163	-
EOTV EQUIV.	1.24	-	41	-	-	-
PROPELLANT	0.864	-	29	-	-	-
TOTAL	-	12	241	72	168	[illegible]

Table 7.3-14. GaAs Dual Sandwich SPS Concept—
TFU Transportation Requirements

	MASS × 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE CONSTR. & MAINT.	22.6	22.6	5.4	99.6	60	3.32	162	100
CREW CONSUMABLES	1.5	0.1		6.4		0.02	6	-
POTV PROPELLANTS	4.3	2.2		19.0		0.32	19	10
EOTV CONSTRUCTION & MAINT.	5.0	-		21.9		-	22	-
EOTV PROPELLANTS	3.8	-		16.8		-	17	-
IOTV PROPELLANTS	0.1	-		0.5		-	1	-
SCB TO GEO	-	-		-		2	-	-
TOTAL	37.3	24.9	5.4	164	60	4	227	110
FLEET	-	-	-	5	5	4	2	2

Table 7.3-15. GaAs Dual Sandwich Concept (MBG)—
Precursor Transportation Requirements

	MASS×10^6 kg	VEHICLE FLIGHTS				
		STS		STS-GROWTH (PLV)	STS-HLLV (CARGO)	EOTV EQUIV GEO X'FER
		PLV	CARGO			
PRECURSOR	3.698	12	136	-	-	1
LEO BASE	5 MODULES	-	-	-	5	-
SCB	14.82	-	-	72	163	-
EOTV EQUIV.	1.24	-	41	-	-	-
PROPELLANT	0.864	-	29	-	-	-
TOTAL	-	12	206	72	168	1

Table 7.3-16. GaAs Dual Sandwich Concept (MBG)—
TFU Transportation Requirements

	MASS × 10^6 kg		VEHICLE FLIGHTS					
							IOTV	
	LEO	GEO	PLV (HLLV)	HLLV	POTV	EOTV	LEO	GEO
SATELLITE CONSTR. & MAINT.	18.0	18.0	5.4	79.4	60	2.64	142	80
CREW CONSUMABLES	1.5	0.1		6.4		0.02	6	-
POTV PROPELLANTS	4.3	2.2		19.0		0.32	19	10
EOTV CONSTRUCTION & MAINT.	5.0	-		21.9		-	22	-
EOTV PROPELLANTS	2.9	-		12.6		-	13	-
IOTV PROPELLANTS	0.1	-		0.5		-	1	-
SCB TO GEO	-	-		-		2	-	-
TOTAL	31.8	20.3	5.4	140	60	3	203	90
FLEET	-	-	-	5	5	3	2	2

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4. Title and Subtitle

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Concept Definition Study (Exhibit D)
Volume III - Transportation Analysis

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16. Abstract

This volume presents the results of a continuing effort to establish the potential impact of the SPS program upon the transportation system concepts envisioned for the SPS construction and operational time period. The results of earlier studies are documented in Rockwell International document SSD 79-0010 (NASA CR-3321).

Additional analyses and investigations were conducted to further define transportation system concepts that will be needed for the developmental and operational phases of an SPS program. To accomplish these objectives, transportation systems such as the Shuttle and its derivatives have been identified; new heavy-lift launch vehicle (HLLV) concepts, cargo and personnel orbital transfer vehicles (EOTV and POTV), and intra-orbit transfer vehicle (IOTV) concepts have been evaluated; and, to a limited degree, the program implications of their operations and costs were assessed. The results of these analyses have been integrated into other elements of the overall SPS concept definition studies.

The primary areas of study during this phase of the contract were directed toward the following: (1) The synthesis and evaluation of a smaller payload version of the HLLV. (2) The assessment of specific technical issues relating to HLLV feasibility. (3) A reassessment of the EOTV concept and configuration update. (4) The identification of technology advancement requirements to enhance/satisfy operations requirements. (5) The generation of cost and programmatic data to support SPS concept trade studies.

SPS program and transportation system analyses continue to show that a prime element of transportation systems cost, and SPS program cost, is that of payload delivery to LEO or HLLV feasibility/cost.

17. Key Words (Suggested by Author(s))

Satellite Power Systems
Launch Vehicle
Electric Orbit Transfer Vehicle (EOTV)
Intra-orbit Transfer Vehicle (IOTV)

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